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STRUCTURAL CONCEPTS AND EXPERIMENTAL CONSIDERATIONS FOR A VERSATILE HIGH-SPEED RESEARCH AIRPLANE

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SUMMARY

Future aircraft may achieve flight speeds to Mach 10 and burn hydrogen fuel. This speed range and cryogenic fuel will impose new demands on the airframe, requiring new materials and structural concepts to result in viable payloads. To enable timely application of advances in structures technology at least risk, large representative structural components will require testing in the increasingly severe flight environments. To accomplish useful flight tests and demonstrations, a versatile, high-speed research airplane has been studied at Langley Research Center. Some objectives of this study are to define a research airplane concept that is capable of testing realistic size components of pertinent future structures at least expenditure of resources.

Reported herein are experimental considerations for a hypersonic research airplane, a means of achieving versatile research capability, a discussion of alternative thermal protection systems, and a definition of the structure and thermal protection system selected for the high-speed research airplane.

The study has identified a near-art structure and thermal protection system and a means of achieving research versatility for a high-speed research airplane. This research airplane is capable of flight testing and/or demonstrating large structural components, hydrogen tanks, and thermal protection systems for a wide range of Mach numbers.

INTRODUCTION

High-speed aircraft of the future may include Mach 6-8 hypersonic transports, Mach 0.8-10 launch vehicles for shuttles, and Mach 4-12 military aircraft. In addition, with the potential of hydrogen fuel, future aircraft requiring further structural research may include speeds ranging from subsonic through supersonic. Numerous advanced structures for these aircraft will require research and development. Consequently, a versatile high-speed research airplane (HSRA) is discussed (ref. 1) that will be capable of flight testing various future structures over a wide range of speeds to Mach 10.

Prior high-speed research airplanes and recent research airplane concepts (ref. 2), had dedicated structure to suit either research airplane goals or one of the operational airplane goals. Effective structural research for numerous types of airplanes requires that a research airplane be capable of testing structures for these future airplanes. Therefore, one objective of the Langley study is to design a versatile research airplane concept capable of testing realistic size components of numerous future structures. Moreover, it is desirable to have a research airplane that can be upgraded to approach prototype construction as technology progresses. To achieve these objectives, an in-house study was performed.

Recent shuttle experience has been applied to the structural design of the HSRA. For instance, the structural arrangement of the HSRA includes an integral tank similar to the integral tank construction proposed by MACDAC (ref. 3) for the HL-10 study. In addition, the heat sink prinicple that studies showed to be economical for the Space Shuttle flyback booster, considered prior to selection of solid rocket boosters, has been adopted for the thermal protection system of the HSRA. At present, however, a more efficient heat sink material

(Beryllium-38 Aluminum) is being studied for the HSRA. Currently, beryllium alloys are available on special order, however, the effect of future changes in health-related requirements related to production and processing is unknown.

The high-speed research airplane structural concept described herein enables flight testing and/or demonstrating of large components made of a variety of structural types and thermal protection systems throughout the speed range of interest. This will be accomplished using replaceable sections thus requiring no modification of the basic airplane structure. As technology develops, the structure can approach prototype construction by incorporating advanced design components for the replaceable payload bay, wings, and thermal protection system. This is the first study of a Mach 8-10 class research airplane to conceive an economical, near-term construction airplane that is designed to enable a broad range of research and development of large structural and propulsive systems.

The paper presents a discussion of a hypersonic research airplane wherein vehicle and structural concepts are presented to illustrate the need for a versatile research airplane, a means of achieving versatile research capability, a discussion of alternative thermal protection systems, and a definition of the structure and thermal protection system selected for the high-speed research airplane. The overall concept and design philosophy for this research airplane are presented in reference 1, and aerodynamic data are given in reference 4.

SYMBOLS

C_p	specific heat
D	cylinder diameter

E	modulus of elasticity
k	thermal conductivity
L	cylinder length
M	Mach number
q	dynamic pressure
T	temperature
ρ	density
σ_{ty}	tensile yield strength

EXPERIMENTAL CONSIDERATIONS

The need to consider structural experiments for the high-speed research airplane stems from the ever increasing severity of environment encountered by advanced vehicles and the increasing complexity of structures required to withstand the environment. Very little flight experience exists with the advanced structural and material concepts. The following sections briefly describe some possible advanced vehicles and structures; discuss the capabilities and limitations of ground testing structures; and present the technical benefits of flight testing and demonstrations.

Advanced Vehicles and structures

Advanced high performance vehicles and structures may include increasingly severe environments and complexities. Figure 1 illustrates several possible advanced vehicles indicating increased Mach number with the accompanying implications of increased aerodynamic heating, and simultaneously these vehicles may be very large and fueled with cryogenic hydrogen. Figure 2 shows the increase in structural concept complexity with increase in flight Mach number. The concepts shown are for liquid-hydrogen-fueled vehicles at a tank location

in the fuselage. As indicated, higher Mach number results in higher aerodynamic surface temperature and higher temperature difference between the aerodynamic surface and the tank wall surface. The higher temperatures require advanced materials, not commonly used in airframe construction. The high temperature difference necessitates consideration of greater thermal stress in advanced structures and means of coping with and alleviating thermal stress. Figure 2 shows three of the numerous structures that may be conceived to satisfy requirements of future vehicles. Appendix A describes several additional concepts to indicate the variety of structures that may require flight testing.

As indicated, material requirements for high performance aircraft are more demanding with increasingly severe environment, larger vehicles and possible use of cryogenic fuel. Figure 3 lists numerous materials for vehicle construction in the future, and different types of structure – hot, insulated and cooled – for a wide range of Mach numbers. Also, many advanced materials are listed including composite materials.

Ground Testing

Ground testing of advanced structures and materials is particularly suited to research and technology development. Numerous concepts can be economically investigated to show feasibility, materials characteristics, and structural behavior. Static tests in the laboratory can be used to verify the predicted general buckling strength of large structural components. Static tests of sufficiently large specimens under simulated flight environments are limited to mechanically applied discrete loads and radiant heating. Factors such as sonic pressure, venting loads, dynamic loads, and flutter particularly

with cryogenic storage are not presently capable of being applied simultaneously with the mechanical and heat loads to large test specimens.

Wind tunnel testing can simultaneously heat and load a structural test model. Depending on design conditions, limit loads may be attained and tunnel tests may be used for determining flutter resistance of a structure or thermal protection system. Current or planned facilities are, however, limited in the ability to simultaneously test specimens of adequate size with representative wall construction in a simulated high-speed flight environment. At high speeds, wind tunnels can test only small structural models or panels. As shown in reference 5, small models with representative wall construction cannot fail in the critical failure mode of general instability; consequently, wind tunnel tests are not suited to structural verification tests of fuselage shells for high-speed aircraft. Wind tunnel tests are suited to airload, heating, and flutter simultaneously with sonic load for short test times.

In general, ground tests are well suited to material characterization tests and element (panel and cylinder, etc.) structural tests. Materials and element tests can be cyclic and for long periods of time. Thus, it is concluded that no ground test facility can simultaneously test large specimens in realistic high-speed flight environment.

Flight Tests and Demonstrations

Research aspects.- Structural research aspects of flight testing include aerodynamic heating to Mach 10 at a dynamic pressure of up to 71.8 kN/m^2 (1500 psf), and aerodynamic loads that may extend to 6 g's to achieve desired structural loading of up to 298 kN/m^2 (1700 lb/in.). Pressure difference around the fuselage is on the order of 0.145 mN/m^2 (1.0 psi), which may act over one or an array of heat shields being tested. For a circumferential array,

ingress of boundary layer air would be a test condition for unsealed heat shields. Panel flutter of heat shields and primary structure panels may also be a research aspect of flight testing. A further aspect of flight testing is it may serve to calibrate wind tunnel results as well as verify predicted aerodynamic loads.

Flight demonstrations.- Flight testing may include large-scale demonstration or proof of concept testing. For instance, a large payload bay structure can buckle by general instability. A smaller specimen cannot fail in general buckling, which is the usual critical failure mode for aircraft. As indicated in reference 6, a cylinder length of from one to three diameters is needed to achieve general instability. The specific length-to-diameter (L/D) ratio depends on the relative longitudinal and circumferential stiffnesses of the cylinder construction. Aircraft structures of the stiffened skin and ring frame construction require an L/D of about 1.5. Transition structures are needed at both ends of a test structure to redistribute loads into the fuselage structure. Consequently, a payload bay length of about two diameters is required to demonstrate fuselage concepts where general instability could be a design concern. Since the HSRA fuselage is about 2.13 m (7 ft.) in diameter at the payload bay, a total payload bay length of about 4.27 m (14 ft.) is needed. Smaller flight specimens are not completely valid demonstrations, since the critical failure mode is not possible. The fact that a large test structure does not buckle in flight at limit loads indicates that the advanced structure is satisfying the requirement in the real environment.

Flight test time required for generation of peak thermal stress in advanced hot structures is shown in figure 4. Test results of a hot structure, reference 5, designed for Mach 8 flight are shown. The results indicate that

the maximum temperature difference occurs during ascent, not at the highest velocity. A maximum thermal stress occurs at about the time of maximum temperature difference. Therefore, short flight times are sufficient to generate peak thermal stresses. In fact, too rapid an acceleration may cause excessive thermal stress, so the ascent trajectory may need to be specifically tailored for some structural tests. Flight testing offers demonstrations of large structural components in the actual environment for useful periods of time.

Operational factors such as rain and runway debris erosion, water entrainment, freezing, and ground handling are only experienced by flight tests. Flight tests require a disciplined development program of the new technology to assure flight safety. Flight tests of large structures to determine service life and to obtain operational experience have been deemed essential to the adoption of composite materials even for the relatively mild subsonic environment.

Flight testing complements ground testing, since ground testing is well suited to static testing of large structures and dynamic wind tunnel testing of small models; whereas, flight testing is suited to demonstration of large structures in the real environment. Consequently, flight testing provides convincing results wherein a demonstration structure having sustained flight limit loads and environment as an integral component of the research airplane can be applied confidently to operational vehicles. Advances in structure and thermal protection for high-speed aircraft may well be critically dependent on the complementing ground and flight testing capability of the future.

RESEARCH VERSATILITY

Research versatility is provided by an extensive range of Mach numbers and dynamic pressures sufficient to include significant portions of flight envelopes of many possible future vehicles. Structural research versatility is provided by modular construction wherein several large sections of the aircraft are replaceable. This feature enables testing large components of advanced structures with representative construction in realistic environment.

Structural Research Goals

The high-speed research airplane structural research goals are to permit a variety of structural research and enable advancements through retrofit so as to approach prototype construction and to achieve this versatility at least expenditure of resources. These goals are to be achieved while satisfying a Mach 10 cruise at a dynamic pressure of 23.9 kN/m^2 (500 psf) for a period of about 1.5 minutes with a B-52 launch constraint. The research airplane is to be capable of testing the structures and thermal protection systems discussed for future operational airplanes. Structural tests are to be sufficiently large to verify the technology readiness for application to future airplanes. Local heating at any location on the airplane is to be accessible for testing specialized TPS as desired. In addition, the lower forebody and aftbody contours are to be variable for scramjet tests.

Flight Envelope

Performance goals are achieved by the flight envelope of the research airplane. Figure 5 shows the flight envelope as functions of altitude and Mach number. The research airplane envelope is bounded by the heavy solid

line, and, as indicated, potential growth extends the performance from Mach 8 to Mach 10, achieved by use of drop tanks.

The flight envelope of the research airplane includes significant portions of flight envelopes for several proposed future aerospace vehicles. Hypersonic transports and airbreathing launch vehicle trajectories are included as well as entry trajectories of spacecraft. With the use of reaction controls, the research airplane can include rocket booster trajectories, thus providing research versatility in terms of applied environment for advanced structures demonstrations for all potential vehicles snown in figure 1.

Replaceable Sections

To enable testing the extensive variety of future structures described in Appendix A, and to perform these tests with adequate size structures, large sections of the research airplane are replaceable (fig. 6). Replaceable sections include the nose cap, payload bay, and wings. In addition, the thermal protection system is replaceable on all surfaces except the elevons and rudder.

A 15' long replaceable payload bay is provided for flight testing realistic fuselage structures and for carrying flight instrumentation and internal flight research payloads. The payload bay length-to-diameter ratio of 2 was based on a general instability criteria for structural failure as described in the section entitled "Flight Testing." Although it is desirable to have a wing sufficiently large to demonstrate resistance to flutter of a test structure, B-52 clearance limits and prior knowledge of the advanced structure preclude this feature as a certainty without further study of demonstration structures. These failure modes (fuselage instability and wing flutter) would enable convincing flight demonstrations, not achievable through tests of small specimens or panels.

Large-Scale Demonstrations

With the large replaceable sections of the high-speed research airplane, an extensive variety of large-scale flight demonstration structures are possible. For example, figure 7 shows an integral liquid hydrogen tank installed as a flight demonstration structure or payload. As indicated, an advanced radiative thermal protection system may be flown simultaneously with the tank structure. Other potential payloads are described in Appendix B to emphasize the need for and value of research versatility provided by the modular construction of the high-speed research airplane.

HSRA STRUCTURAL DESIGN

The above discussions indicate a variety of structures will be needed for future airplanes and flight testing will be warranted to advance the structures art. The rationale for structural design of the high speed research airplane, to enable effective flight testing of future structures, are given in the following sections.

Alternative Concepts

Several alternative concepts are available for the basic HSRA structure and thermal protection system. Some alternatives can offer improved performance through reduced weight. However, the more efficient alternatives generally have less life or require more development than the selected baseline concepts described in subsequent sections. Comparative performance and cost analysis are needed to enable final selection of structural alternatives for the HSRA. Alternative concepts considered for the HSRA are described in the following subsections.

Heat sink shields.- Heat sink shields allow research versatility by providing the stand-off thermal protection feature (as seen in figure 8(a)) at low cost since the heat sink concept is a well established approach. Heat shields made as heat sinks offer simplicity since they can be operated at a moderate temperature. No insulation is needed between the shields and structure; and the development of high temperature seals for edges of slip-jointed shields is avoided. The relatively thick shields are stiff, thus requiring little, if any, added stiffening while offering rigid edges suited to sealing against inflow of hot boundary-layer air.

Moderate temperature heat sink materials include aluminum alloys and beryllium-aluminum alloys. Pure beryllium and nickel-base alloys may be operated to higher temperatures; however, packaged insulation (fig. 8(b)) is needed to protect an aluminum alloy primary structure. And the sealing problem is more difficult with higher temperature shields.

Aluminum has been considered for heat sink shields because aluminum has been shown in studies to be cost effective as a heat-sink structure for the shuttle flyback booster. However, as seen in figure 9, aluminum is not as efficient as (Be-38Al) a beryllium-aluminum alloy known as Lockalloy. Lockalloy has more than twice the specific heat and about twice the allowable temperature rise of aluminum. Analyses indicate that aluminum shields require more than three times the mass as Lockalloy shields and, therefore, seriously detract from flight performance.

Nickel alloy as a heat sink material for shields is less effective than operating at a higher temperature where most aerodynamic heating is radiated from the shield. Newer superalloys allow radiation equilibrium temperature operation for a wide range of Mach numbers. The Inconel X of the heat-sink

structure of the X-15 was limited to about 922°K (1200°F). At this temperature the problem of insulation and sealing exists at a weight penalty over radiative thermal protection designed to operate at radiation equilibrium temperature.

Beryllium, which has been used for shields on the Gemini parachute canister, holds promise for improved flight performance. Beryllium has a higher specific heat than Lockalloy, thus beryllium can be lighter for the same performance. In addition, beryllium has a higher operating temperature, which when coupled with relatively near term insulation packages under the shields (fig. 8(b)), provides a flight performance gain. The gain is significant, see Table II, warranting consideration of beryllium heat sink heat shields. As indicated in Table II, at a mass of 2722 kg (6000 lbm), beryllium offers the same performance as 3175 kg (7000 lbm) of Lockalloy for the same temperature rise of 589°K (600°F). Upgrading the 2722 kg (6000 lbm) beryllium system by adding 680 kg (1500 lbm) of insulation and packages and allowing a beryllium temperature rise of 811°K (1000°F) offers 1.3 minutes of cruise time at Mach 10 and at a dynamic pressure of 71.8 kN/m^2 (1500 psf); whereas, the 3175 kg (7000 lbm) and 589°K (600°F) Lockalloy system cannot achieve Mach 10 (M=9.3 max) at a dynamic pressure of 71.8 kN/m^2 (1500 psf). Study of beryllium mechanical and fabrication properties are needed to determine whether or not beryllium is better than Lockalloy. Lockalloy has better ductility than beryllium, but the ductility of beryllium may be adequate for heat sink heat shields.

Reusable external insulation.- Carrier panels, shown in figure 10(a), may replace the baseline heat shields wherein the carrier panels are covered with reusable external insulation. Present shuttle development of RSI may result in a rigidized fibrous insulation that has an adequate cycle life. Further

development may be necessary for application to the HSRA. Attachment of the carrier panels may be better provided by continuous support along the leading and trailing edges rather than the post supports of the heat sink shields. Continuous support may be provided by ring frames attached to the outside of the primary structure on the fuselage and by z-section stringers attached to the outside of the spar caps on the wings. Carrier panel seals and RSI attachment to the carrier panels are developments that may differ from shuttle technology therefore requiring further development for HSRA application.

Ablation.- Ablation may be a candidate TPS for HSRA particularly if the RSI materials prove to be inadequate for HSRA application and if less weight is necessary than required for heat sink shields. Refurbishment costs may be minimized by replaceable carrier panels, but initial cost and weight will increase. The carrier panel, shown in figure 10(b), is simply removed and replaced after each flight. Low density ablators of the type flight tested on the X-15 appear suited; however, many surfaces may benefit from an ablator that has a pyrolysis temperature approaching or greater than the radiation equilibrium temperature thereby reducing refurbishment requirements. However, the fragile nature of the relatively brittle char layer that forms where pyrolysis occurs requires further study of refurbishment requirements.

Elementary radiative TPS.- An elementary radiative thermal protection system is shown in figure 10(c). A relatively simple metallic heat shield of flanged waffle plate construction is fastened to nickel alloy standoffs. High temperature insulation is enclosed in a sheetmetal container. The container supports venting pressure differences by three longitudinal beams. A center web and the enclosure sides form the three beams. One required

development is high temperature sealing of heat-shield edges. To reduce the effects of leakage, the insulation packages have overlapped edges to provide an additional seal. The space between the shields and the insulation packages is vented to help limit lateral flow of leakage to only this space.

Heat-sink structure.- Heat-sink structures (fig. 11(a)) have been considered for the high-speed research airplane. However, research versatility is reduced because advanced thermal protection concepts cannot be tested as readily or as realistically as with the standoff TPS selected for the HSRA. Removal of heat sink panels and replacement by an equivalent submerged structure covered with an advanced TPS does permit experiment at essentially any place on the vehicle. However, ring frames, ribs, and spars restrict the size and position of test panels. Moreover, the cavity pressure and lateral flow conditions in a cavity are not those of a fully heat-shielded structure, thus heat-shield leakage conditions are not realistic.

A slip joint is provided at the wing root, thus a hot heat-sink structure of nickel alloy, similar to the X-15, may satisfy HSRA wing structure requirements. Higher temperature materials, such as titanium and nickel base alloys, are possible heat-sink structural materials, but Lockalloy has a higher thermal efficiency except for systems that approach radiation equilibrium temperature, i.e., hot structure. Lockalloy is a new technology, developed after the X-15 and promises simplified HSRA TPS and structure.

Cost considerations favor a heat sink structure over a stand-off shielded structure, so heat-sink structure for the HSRA warrants further assessment.

Hot structure.- Hot structure (fig. 11(b)), which operates at radiation equilibrium temperature, is an alternative that was proposed for a delta wing for the X-15. Hot structure operating at high temperature is subject to creep-

induced residual stress and significant thermal stress. Moreover, fabrication with nickel alloys is considerably more costly than with lower temperature metals. However, hot structure has received more study than other concepts, and, therefore, may satisfy the HSRA structural requirements provided it can be shown to be weight and cost effective.

Direct bond insulators.- Insulating materials bonded directly to the structure (fig. 11(c)), such as the reusable external insulation being developed for the shuttle, offer a possible low weight alternative. However, like the heat sink structure, research capability is not as versatile as with a stand-off thermal protection system. A further consideration is that the fragile nature of the external insulators poses operational difficulties for a research airplane. Numerous access doors and structural replacements will require considerable handling of the vehicle surface. A brittle material is more subject to damage than a metallic surface such as a heat sink TPS.

Nonintegral tanks.- With a stand-off thermal protection system an integral tank has higher volumetric efficiency than a nonintegral tank; volumetric efficiency is a critical criteria for HSRA performance. The selection of storable propellants for the basic HSRA avoids cryogenic insulation development and possible thermal stress problems of integral tanks associated with cryogenic propellants. The large payload bay of the HSRA provides the opportunity to perform research on cryogenic tank systems of either integral or nonintegral construction. However, nonintegral tanks for the basic HSRA propellants may offer more flexibility in choice of propellants for various research engine options. Integral tanks may be compartmentalized to provide storage of different propellants for research with different propulsion systems (ref. 4); however, cryogenic storage in integral tanks would depend on development of reusable insulation. Selection of nonintegral tanks

for HSRA would permit research at less development with both storable and cryogenic propellants with some loss in tank volume with the stand-off TPS. A study is needed that determines what propulsion research and related propellants are to be tested on the HSRA and which tank arrangement, integral or nonintegral, best satisfies performance and flexibility goals.

HSRA Structural Concepts

HSRA structural concepts based on approaches dedicated to only a research airplane, such as the hot heat sink structure of the X-15, offer lowest cost structures. Structural concepts dedicated to only an operational airplane, such as an advanced radiative TPS on a boron-aluminum primary structure are highest cost structures. The approach selected for the basic HSRA is a combination of these two approaches, and is shown in figure 12. A more recent technology, offering better thermal efficiency than the X-15 nickel-alloy heat sink structure, has been applied to heat shields for the HSRA. Lockalloy (Be-38Al) heat sink shields are attached by standoffs to the structure. The structure is a state-of-the-art aluminum alloy skin-stringer construction. Integral tanks (figs. 12a, 12b, and 12d) with partitions allow research of different engines and fuels. The conventional structure and near-term heat sink thermal protection offer low cost. Research versatility is provided by a large replaceable payload bay structure in the fuselage that has a length of twice its diameter, by replaceable wings, and by a replaceable thermal protection system for the entire airplane except for elevons and rudder. Moreover, a space is provided between the heat shields and structure for versatility. The Lockalloy shields permit flight performance that satisfies the objectives. At speeds lower than Mach 10, the heat sink

system offers extended cruise times. As seen in figure 13, the HSRA can cruise 1.4 minutes at Mach 10 and at a dynamic pressure of 23.5 kN/m^2 (500 psf). Further, as the Mach number is reduced, the cruise time is increased. As shown for Mach 4.5, 10 minutes of cruise time are available. To make the heat-sink thermal protection system most effective, minimum heat load ascent and return trajectories were selected.

The heat-sink principle has been selected for the leading edges for the same reasons this principle was selected for the basic thermal protection system. Analyses performed to date show the leading edge weight is acceptable in Lockalloy. However, these analyses assumed uniform temperature throughout the leading edge chordal length and they did not include shock impingement or yaw effects on heating rates. A more detailed analysis is needed to determine if the higher permissible operating temperature of beryllium warrants its consideration for the leading edges, or if some other leading edge concept may be better for the basic airplane. However, the lower temperature of the Lockalloy leading edge simplifies the slip joint sealing problem which has proven to be critical for refractory-metal leading edges.

Primary Structure

The primary structure of the high-speed research airplane fuselage is shown in figure 12. Thrust from the five rocket motors is supported by a rectangular grid of beams that form the thrust structure bulkhead. A longeron is attached to each end of each beam to dissipate the concentrated thrust loads over the aft fuselage shell. Longerons terminate at the aft payload-bay bulkhead.

Storable propellants are contained by integral tanks. The tank cross section conforms with the fuselage cross section except at tunnels and wheel

wells. Transverse tension webs are attached to each ring frame on about 0.51 m (20 inch) centers to support internal pressure. Each tension web has a ring reinforced access hole. Tank walls support pressure by bending between frames. Wall panels have flanged integral stiffeners oriented longitudinally and external to the tank skin. Panels are fusion welded at thickened weld lands. Similarly, the tank bulkheads have machined faces with thick weld lands for reliable sealing. The face of the bulkhead separating the oxidizer and fuel is double-wall construction for fail-safe design. Flat bulkheads provide maximum volumetric efficiency. Volumetric efficiency is a critical design criteria for B-52 launch constraints and for armpit mounted research engine capability. The B-52 constricts the body height for ground clearance and wing pylon position and the armpit engines constrict body width.

Primary structure for the payload bay and pilot equipment and engine compartments are made of riveted sheet metal construction. Longitudinal z-stiffeners are fastened externally to the skin with internal frames.

The wing structure, shown in figure 14, consists of spanwise oriented z-stiffeners riveted externally to the skins. A flat lower wing surface offers simple panels. A curved upper surface was selected instead of flat areas and high points to avoid high point spars which intersect many panels, ribs and spars imposing an excessive number of parts. Ribs and spars are riveted sheet construction with extruded T-section caps. Webs are stiffened by z-stiffeners for compression support between upper and lower caps. Between the z-stiffeners the webs have flange reinforced lightening holes. Adjacent to the leading edges, full depth bonded aluminum honeycomb-core sandwich replaces the z-stiffened skins as well as the ribs and spars.

Control surfaces, shown in figure 15, have fabricated ribs and spars riveted to thick Lockalloy face sheets. The face sheets are made of Lockalloy to serve as heat sink structure. This construction permits full depth structure for thin control surfaces while providing the same thermal performance as the basic HSRA thermal protection system.

The wing structure is attached to the body structure, as shown in figure 12(d), at each of the spars, which have the same spacing as the body frames. The wing is pin jointed at upper and lower spar caps to corresponding beam caps in the fuselage. All pins are parallel to the flat lower wing surface to permit unrestrained thermal expansion by slip on the pins for a hot structure wing research option. Fuselage beams that provide wing carry-through structure are the upper section of the fuselage frames.

Thermal Protection System

Thermal protection for the basic HSRA, as shown in figures 12 and 16, consists of waffled and formed slabs of Lockalloy. These heat sink shields vary in thickness from about 1.27 cm (0.5 inch) to 2.54 cm (1.0 inch) and are about 0.51 m (20 inches) by 0.51 m (20 inches) square. Each shield is slip jointed along its edges. Heat shields on the upper surfaces and body sides are no less than 1.27 cm (0.5 inch) thick although heat sink requirements call for less than this thickness. In these areas, the shields are waffled to approach the unit mass required for heat sink and to provide stiffness and depth required for flutter resistance and load support. Shields are individually supported by shared titanium standoffs attached to the structure at ring frames and spars. Slip joints, shown in figure 16, are sealed to prevent inflow of hot boundary layer air by Teflon gaskets. The maximum shield temperature is 570°K (600°F). Overlapping shield edges

are bolted to one another on about 7.67 cm (3 inch) centers to provide a positive seal. Each bolt has an enlarged or elongated hole in the inner shield to permit unrestrained thermal expansion. Slip joints are also provided for each shield at three of the four support posts by slotted holes at all but one corner of each shield. Space between the shields and primary structure is provided to permit installation of packaged insulation for an advanced radiative TPS research package. Wall thicknesses of 15.2 cm (6 inches), 12.7 cm (5 inches), and 10.2 cm (4 inches) are provided for lower, side and upper surfaces, respectively, of the fuselage. Lower wing wall thickness is 10.2 cm (4 inches) and upper wing wall thickness is 7.62 cm (3 inches). A smaller wall thickness is provided for the wing than for corresponding fuselage surfaces to permit a maximum wing structure thickness of no less than 2.0 percent at any wing station. These structural thicknesses along with the reduced wall thicknesses result in maximum wing mold-line thicknesses of about 7 percent at the tip and 3 percent at the body centerline. A denser insulation than optimum may be needed for the wings with advanced radiative TPS demonstrations. The body wall thickness should be sufficient for optimum insulation density for an advanced radiative TPS research package.

Both the nose cap and leading edges are made of Lockalloy. Like the heat shields, beryllium may be a better material for the leading edges. Leading edges are segmented and slip jointed. Each segment (except end segments) is identical for interchangeability and reduced cost. The cross section of the leading edge is constant for the entire leading edge. The cross section is defined by the tip airfoil. The forward upper wing contour is blended to suit the constant leading edge shape and the aft upper wing surface contour.

Structure and TPS Weights

Performance analyses reported in reference 1 were made on the basis of estimated weights for structure of the HSRA. Thermal protection weights were calculated based on ascent and descent trajectories of minimum heat load. Later analyses included stress analyses of the structure to provide a weight estimate based on loads. Table I lists the initial structural mass estimates and the calculated values. In addition, the landing gear weight was initially estimated and later analyzed. The estimated structure plus landing gear mass is 3358 kg (7404 lbm), and the calculated value is 3374 kg (7439 lbm). This close agreement was not realized for each component; however, the performance analysis is affected only by the total weight. The thermal protection mass is 3175 kg (7000 lbm). Both the structure and thermal protection require detailed design, analysis, and optimization to define the design weight. A modified B-52 launch vehicle could support a mass of 49,900 kg (110,000 lbm); whereas, the basic HSRA with drop tanks is estimated to have a mass of only 42,200 kg (93,000 lbm). Thus, a weight growth is permissible without effecting program goals.

Analytical Basis for Weight Estimates

This section presents a brief review of the analytical basis for the structural and thermal protection system weight estimates of the HSRA.

Wing analyses are for the limit load symmetrical 2.5g pull-up maneuver performed after release from the B-52 carrier aircraft. Sixty percent of the lift is assumed for the exposed wing -- the remainder for the fuselage. A gross mass of 27.2 kg (60,000 lbm) was assumed. This is the greatest wing load condition. The HSRA has a greater gross mass when flown with drop tanks, but the drop tanks are attached to the wing, thus producing a load alleviation

for the wing. Therefore, the drop tank loads were not considered. The subsonic lift distribution was considered to be centered on the 25 percent chord line. Lift forces were distributed accordingly for each of the 50.8 cm x 50.8 cm (20 inch x 20 inch) wing panels. For each spar, spaced on 50.8 cm (20 inch) centers, shear and bending moments were determined for the airloads acting on the corresponding wing panels. With known wing thickness at any location, spar caps were selected adequate for panel attachment and with the required shear web the caps form I-beam spars of known dimensions. The bending moment capability of the spar was determined at various span locations. Local buckling of the cap was the critical design consideration. The total ultimate bending moment, applied over a 50.8 cm (20 inch) wide strip, was reduced by the bending moment capability of the spar. The remaining bending moment was carried by the spanwise oriented Z-stiffeners, and the skin. The local buckling of the Z-stiffener elements and skin between Z-stiffeners were set about equal to one another and equal to the column buckling strength of the Z-stiffener-skin conbination. The local buckling strength of the spar caps was adjusted to equal that of the wing panels. Upper and lower surface panels were made identical in cross section or equivalent thickness. The above approach was used for each wing panel and spar. Ribs, elevons, and wing-body attachment fittings were analyzed in a similar manner to enable calculation of total wing weight.

A NASTRAN analysis was performed on the wing to verify the weight estimate based on the stress analysis discussed above. Figure 17 shows the NASTRAN model generated for the upper surface of the wing structure. A similar model was made for the lower surface, and each surface was joined by rib and spar webs. Iterative analysis yielded a weight close to that estimated by the stress

analysis method. The NASTRAN analysis also provides wing reactions (fig. 18). These loads are indicative of those that have to be supported by a research wing structure. Spanwise edge-compression load intensities of up to 690 kN/m (3945 lbf/in) must be supported by the wing panels. This maximum load intensity covers the range of load intensities predicted for large hypersonic vehicles.

The fuselage analysis is based on the same 2.5 g pull-up maneuver for the wing. The air load was distributed equally over the planform body area, and wing shears were added at respective body attachments. Shears and bending moments were determined from the net air and inertial loads along the body length. In the upper integral tank panels, a beam-column analysis was performed. The axial load from bending was combined with the lateral 68.9 kN/m^2 (10 psi) tank ultimate pressure load. Tank side panels were analyzed in a similar manner, but half the body bending moment used for upper panels was assumed. Internal tension webs were sized to support pressure force acting on opposite body halves. A reinforcing ring weight of twice the weight of the metal removed by the access hole was used. Other tank structure such as rings and bulkheads were detailed on design drawings, which were used to estimate weight of all parts.

The forward fuselage had external z-stiffeners and internal rings. It was not pressurized, so only axial load from body bending was considered in the analysis. As for the wing, local and column buckling stresses were made to be equal by iteration of element sizes, and this stress was equal to the ultimate applied stress. Ring weight and proportions were arbitrary, but judged to be sufficient to preclude general instability. An optimization analysis of the body would be required to minimize body weight, but such analysis was beyond the scope of the study. However, a NASTRAN analysis was

performed for the payload bay structure. Weight derived from the NASTRAN analysis was about equal to that derived by the methods discussed above. Figure 19 shows a cross section of the payload bay structural model generated for NASTRAN. Reactions at the attachment of the payload bay to the integral tank structure are given in figure 20. These reactions are indicative of the loads that must be supported by a demonstrator payload-bay structure. A peak edge-compression load intensity of 115.9 kN/m (662 lbf/in) must be sustained at limit load by the payload bay panels. This maximum load intensity is within the range predicted for large hypersonic vehicles (ref. 5).

Design of the main landing gear was performed to assure that the gear was suited to both bottom-mounted scramjet and armpit-mounted ramjet research packages. The resulting struts were analyzed for strength, and made from 4340 M heat-treated steel. Limit load was 2.0 g and ultimate load was 3.0 g ground taxi condition. This load condition was selected for initial sizing of the landing gear.

The thrust structure bulkhead, engine compartment, and vertical stabilizer were all detailed on design drawings. Weight estimates were made through engineering analysis and weight calculations made for other parts of the aircraft.

The heat sink mass was calculated based on a variety of trajectories that have minimum heat loads. A range of Mach numbers and dynamic pressures were analyzed up to Mach 10 and 71.8 kN/m^2 (1500 lbf/ft^2). Resulting Lockalloy weights are shown in figure 21 for a dynamic pressure of 23.9 kN/m^2 (500 lbf/ft^2). As indicated, the 3175 kg (7000 lbm) of Lockalloy offers 1.4 minutes of cruise time at Mach 10. This same Lockalloy mass offers over 4 minutes of cruise time at Mach 8.

CONCLUSIONS

A Langley study of a high speed research airplane (HSRA) has progressed through conceptual design. This study indicates the following: a versatile high-performance research airplane is achievable without structural modification over a wide range of Mach numbers and can be fabricated by near-term technology requiring minimal RDT&E effort. An assessment of future vehicles covering the speed range of subsonic through hypersonic indicates that there are numerous possible advanced structural and propulsive systems warranting large scale flight testing.

Versatile flight testing is provided by a replaceable fuselage section or payload bay of sufficient size to flight demonstrate realistic fuselage structures. The payload bay was designed with a length-to-diameter ratio of two to satisfy a minimum size criteria for general instability of aircraft wall construction. Moreover, the wings are replaceable and have a slip-joint type attachment to the body to enable tests of hot structure wings. The entire thermal protection system is removable and space is provided between the thermal protection and the structure. Thus, flight tests of advanced thermal protection systems are possible for the entire airplane surface or for any local area of interest. As technology is developed the payload bay section, wings, and the entire thermal protection may be replaced by advanced constructions to approach prototype construction throughout. The contour of the outer mold line may be modified by reconstructing thermal protection shields to investigate, for example, the inlet or exhaust flow field of a scramjet research engine. Versatility is further provided for engine testing in that the fuel tank is compartmentalized to permit storage of varying amounts of different fuels. Moreover,

hard point attachments are built into the structure for mounting research engines and for mounting research liquid hydrogen tanks.

With a B-52 launch and with drop tanks similar to the X-15A2 research airplane, Mach 10 cruise at a dynamic pressure of 23.9 kN/m^2 (500 psf) is provided for a period of about 1.5 minutes. At a Mach number of 4.5, 10 minutes of cruise time are available. The available cruise times are sufficient for both advanced propulsion and structural research. These cruise time Mach number combinations are made possible through a heat sink thermal protection system that does not exceed 589°K (600°F) for any mission to speeds up to Mach 10. This low temperature simplifies the sealing of heat shields and negates development of packaged insulation and lightweight shields for the research airplane. This is particularly attractive when consideration is given to material selection as a function of Mach number for radiative thermal protection. The basic research airplane thermal protection system consist of relatively simple 50.8 cm (20-inch) square slabs of beryllium-aluminum alloy (Lockalloy) mounted by titanium standoffs to the underlying aluminum structure.

The aluminum primary structure is a state-of-the-art skin-stringer construction. An integral propellant tank efficiently contains the low pressure storable propellants.

The subject study has identified several possible research payloads that consist of advanced construction for the replaceable sections of the airplane These research payloads can provide a focus for technology development which may culminate in realistic flight testing of the better alternative solutions. Flight-tested structures serve to man-rate the structures, and their development is enhanced through the required

disciplined research required for flight safety. Performance gain potential of new technologies can be probed, and operational experience attained. Moreover, the RDT&E costs of future airplanes should be reduced through the research airplane test program which can provide effective design options for structures and propulsive systems for many possible future vehicles.

APPENDIX A

FUTURE VEHICLE STRUCTURES

With the possible advent of hydrogen fuel, larger airplanes and higher flight speeds, new structural-material concepts would be needed to withstand the environments and offer viable payloads. A discussion of some possible future structures is presented below for various airplanes to indicate the need for versatility in structural research capability of a research airplane. A number of structural concepts are discussed for vehicles that range from subsonic through hypersonic speeds. However, hydrogen tank systems are common to all speed ranges for hydrogen-fueled vehicles, so initial discussion pertains to hydrogen tank systems.

Aircraft Hydrogen Tank Systems

Aircraft hydrogen tank systems are of interest because increasing difficulty in satisfying demand for petroleum fuel may require a change in fuel types. One fuel particularly suited to aircraft is hydrogen. The high specific energy enables longer ranges and the products of combustion are relatively free of pollutants. In addition, high-speed aircraft can benefit from the enormous cooling capacity of hydrogen fuel.

Tank construction.- Efficient storage of hydrogen is provided through liquefaction. However, the density of liquid hydrogen is low requiring four times the volume of kerosene for the same energy. Consequently, high volumetric efficiency is desirable for hydrogen tanks. Figures 22 and 23 illustrate potential tank designs offering high volumetric efficiency for storage in compartments that are not circular. Multi-lobed tanks (fig. 22)

can effectively fill a noncircular cavity; and noncircular tanks (fig. 23) can offer further improvement in volumetric efficiency for a given thickness of insulation.

Generally, for storage spaces that permit use of the full cross-section of the fuselage for fuel, tanks that also serve as the structure (integral tanks) offer a further improvement in volumetric efficiency. The combining of fuel storage and structural functions potentially saves weight. However, life considerations may result in increased integral tank weight since nonintegral tanks may be designed for less life and be replaced during service. Integral tanks employ internal insulation to minimize thermal stress and low temperature effects on the tank material. Because of hydrogen diffusion into the insulation, available internal insulations have an order of magnitude greater thermal conductivity than similar insulations applied to the outside surface of the tank. Further study of the above considerations is required to determine whether or not potential weight, cost and volumetric efficiency advantages of the integral tank can be achieved.

Cryogenic insulation.- Liquid hydrogen is extremely cold and relatively volatile. The low temperature can result in high heat transfer to the fuel even with ambient sea-level environment. Moreover, an uninsulated liquid hydrogen tank can condense air at a rate sufficient to boil off 18 pounds of fuel per hour for one square foot of tank surface area (26.78 kg/m^2), (ref. 7). Therefore, hydrogen tanks must be well insulated and air must be kept from condensing.

Cryogenic insulations for application to the outside of tanks are shown in figure 24. The illustration is for nonintegral tanks wherein the insulation is protected from erosion by the primary structure. For integral

tanks, the insulation may be protected by a fiberglass cover (ref. 8) for subsonic aircraft and by heat shields of composites and metals for high speed aircraft.

With a porous fibrous insulation, the cold tank wall can condense any gas except helium. The most developed TPS for cryogenic tanks (ref. 9) used helium purging of the insulation space surrounding the tank as shown in the upper left of figure 24. However, the high thermal conductivity and cost of helium provided incentive to improve cryogenic insulation systems. Saturn SII stages are insulated with closed cell foam. This foam fails acceptably during ascent, but is not suited for reuse. A more recent development is a fiber reinforced closed cell foam (ref. 10). However, the number of reuse cycles is not known and this polyurethane foam is temperature limited to 353°K (175°F). For higher temperature applications, the polyimides and pyronnes, for example, require study to attain a reusable closed cell foam that is impervious to a purge gas such as nitrogen.

In the CO_2 purge system (ref. 11) shown at the lower left of figure 24, CO_2 frost is cryodeposited with the aid of helium during preflight. Sublimation of CO_2 absorbs heat that may transfer to the fuel and provides in-flight inert purge gas. However, the CO_2 purge system requires some helium and imposes an operational constraint in that a known quantity and density of frost must be deposited before flight.

Evacuated multilayer foil (super insulation) (ref. 12) is shown at the lower right of figure 24. Even with a pressure load acting on the insulation, it is more efficient than the other cryogenic insulations. However, sealing the jackets and the joints between jackets presents reuse difficulties.

Internal cryogenic insulations are also emerging technologies, and are shown in figure 25. The Saturn S-IV-B stage had a three-dimensional fiber reinforcement of blocks of polyurethane foam. An inner surface of sealed fiberglass cloth is added to prevent penetration by liquid hydrogen. A more recent development is a gas film insulation that consists either of open-ended organic tubes bonded to the tank wall or of honeycomb bonded to the tank and covered with a perforated Mylar facing (ref. 13). Liquid hydrogen penetrates into the core, gasifies and equalizes pressure over the miniscus at the perforations preventing further liquid penetration. Each of these internal cryogenic insulations has a relatively high conductivity since each is filled with hydrogen gas. Impermeable liners and life tests are possible research goals for internal cryogenic insulations.

Cryogenic fuel safety.- Safety aspects of hydrogen storage require reliable tank sealing and possibly purging of the enclosure around the tanks. A tank material showing promise of reliable storage is 2219-T87 aluminum alloy. Welds are made of thickened lands that are typically 0.95 cm (0.375 inch) thick to minimize leakage. Stiffeners if required are integrally machined to avoid fastener penetrations of the tank skin. Access doors may require shear seals to meet acceptable leak rates; however, the Teflon gaskets with close bolt spacings, as developed in reference 14, appear suitable for reusable seals warranting further testing.

Inert purging of enclosures containing tanks can provide safety. Onboard inert purge systems are not available, and are a subject for development. Actually, a design criteria is needed on purging, since it may be possible to safe the tank enclosures by purging with air, permitted for ground-based equipment. Flight testing can prove the operational aspects of safety considerations for hydrogen storage in aircraft.

Subsonic Aircraft

For subsonic aircraft, the advanced structural concepts will be concerned with such factors as composite materials, load alleviation controls, configurations to distribute loads and lift forces to reduce internal loads, and the tank alternatives described in the previous section. Integral tanks may evolve, but for subsonic airplanes the integral tank wall may serve as the outer mold line, as shown in figure 26; therefore, added complexity results since the rings and stiffeners must be insulated within the tank. Internal joints will not be readily accessible for the necessary periodic fracture control inspections. Operational experience is needed to assess many of the above structural advances for subsonic airplanes. A research airplane is capable of resolving the hydrogen storage problems at minimum risk since hydrogen fuel is considered in the initial design of the airplane rather than as a retrofit to an existing transport. Moreover, this research airplane can apply greater flight loads to the tank system than possible with a similar size tank in a converted transport. Therefore, tank development could benefit by a research airplane followed by a converted transport demonstrator.

Supersonic Aircraft

Supersonic aircraft performance is enhanced by hydrogen fuel. The increased efficiency of hydrogen may enable low sonic boom airplanes that have greatly improved environmental aspects. However, the aerodynamic heating imposes new requirements on the structure.

Studies to date of Mach 3 class supersonic transports have taken the hot structure approach, an example of which is shown in figure 27. The Ti-6Al-4V alloy is the primary material choice, which is well suited to the Mach 3

environment. Newer titanium alloys and the boron-aluminum composites (with some external thermal protection) appear suited to Mach 4.5 environment. Hydrogen tanks are made of 2219-T87 aluminum alloy as with subsonic airplanes. Integral tanks are protected structures in that packaged higher temperature insulation is needed in addition to the cryogenic insulation as shown in figure 28. Heat shields protect the insulations, and the shields may be either metallic or nonmetallic. Titanium and the polyimide-graphite composite are respective candidates for shields.

The high heat capacity of hydrogen enables cooling the airplane structure as well as the engines. Active cooling (ref. 15) may permit use of aluminum structure, as shown in figure 29, or low temperature epoxy-graphite composite instead of the more costly materials used for hot structures. Moreover, a weight savings may result by cooling the structure.

Hypersonic Aircraft

Hypersonic aircraft studies show several approaches to structural design are possible. These include hot structure, partially insulated structure, insulated structure, and actively cooled structure. Each approach is a subject for research and possibly flight testing.

Hot structure.- An extension to hypersonic aircraft of hot structure in nickel-base alloys, such as Inconel 718 for Mach 6 and Rene 41 for Mach 8, has been studied and shown in figure 30. A tubular stiffened panel structure of Rene 41 is shown for a hydrogen tank compartment. Such semimonocoque structures are suited to hot structure since the beaded skins offer thermal stress relief through low transverse inplane extensional stiffness as indicated in reference 16.

Statically determinate structures also have been considered for hot structures to avoid thermal stress; however, the reduction in thermal stress

is the result of reduced stiffness requiring added weight to avoid flutter. Consequently, semimonocoque and fully monocoque structures such as honeycomb-core sandwich have been shown to offer lighter hot structures.

An unresolved problem area for hot structures is creep induced residual stress. Creep will occur only at the hotter, higher stressed areas; therefore, the creep will not be uniform throughout the structure. Nonuniform creep will result in local permanent strain in the structure which when cool and unloaded will be forced to return to nearly the original geometry by the bulk of the structure which experienced no creep. The result will be residual stress which varies with time at temperature rendering a complex stress-versus-time history. Possible solutions are to insulate the hotter areas (to reduce the structural temperature and to reduce temperature gradients) and to actively cool the structure.

Partially insulated structure.- For areas of high heating rate, partially insulated structure (fig. 31) is beneficial. Partial insulating limits the maximum temperature of the structure for efficient load support and for control of temperature gradients to reduce the thermal stress. Lowering the structure temperature also avoids creep and creep-induced residual stress. By control of the temperature gradients, the rigid semimonocoque and fully monocoque structures are more efficient than the flexible statically determinate structure as shown in ref. 15.

Insulated structure.- Insulated structure limited to low operating temperatures of conventional aluminum alloy or low temperature composites, shown in figure 32, reduces thermal stress and temperature related material problems. Considerable insulation weight and thickness are required and the entire aerodynamic surface is shielded. Shield design is a problem common

to most structural concepts since some shielded areas are usually required. Insulation weight, however, is greater for the cool structure design than for either hot structure or actively cooled structure. Phase change materials (ref. 17) contained and attached to the cool side of the insulation package show promise of reducing both total thermal protection system weight and insulation thickness for insulated structures.

Actively cooled structure.- Actively cooled structure of aluminum alloy and possibly of metallic composites offers promise of less weight than either hot or insulated cool structure. In the actively cooled concept (fig. 33) proposed in reference 18, weight is reduced by using all of the available heat sink capacity of the fuel to cool over two-thirds of the aerodynamic surface without external thermal protection. In these areas the aluminum structure is exposed to the hypersonic environment. Fail-safe devices and failure detection are subjects of research for actively cooled structure. Among these devic are redundant cooling circuits, heat sink skins and radiative thermal protection. Actively cooled structure as well as the hot and insulated structures have yet to be optimized and compared, so the flight test capability of a research airplane should be sufficiently versatile to test any type of structure.

Airbreathing Launch Vehicles

Operational cost and launch site flexibility are enhanced by a reusable launch vehicle for the space shuttle. A rocket-powered winged booster was studied in the shuttle program. These studies indicated that an aluminum heat sink structure is cost effective. However, with a reusable booster, airbreathing propulsion offers improved shuttle performance. Airbreathing launch vehicles may be hypersonic and also benefit from hydrogen fuel. Insulated

hot and actively cooled structure are all contenders for airbreathing launch vehicles. In many respects the problems in particular hypersonic airplanes are the same as for hydrogen-fueled airplanes. Exceptions may be that reuse life and weight may not be as critical as for a hypersonic transport. A lower fabrication cost approach may be desired even with a weight penalty. With a reduced reuse requirement, the external insulation being developed for the space shuttle may be directly bonded to an aluminum structure, as shown in figure 34. However, without refurbishment, a Lockalloy (Be 38Al) heat sink structure may significantly reduce booster weight and size which may lower cost from the aluminum heat sink booster. With an increase in weight, a relatively simple radiative thermal protection system may be suitable. For instance, machined titanium and nickel alloy shields may be simpler than the lighter weight, sheet-metal sandwich shields. Flight testing airbreathing launch vehicle structure is ideally suited to the HSRA which has about the same flight environment.

Military Aircraft

Military aircraft not only may include most of the LH_2-fueled aircraft types discussed above, but include high-speed petroleum-fueled airplanes as well. The upper speed for petroleum fuel is limited by engine cooling to about Mach 4.5. This Mach number environment imposes the upper temperature limit of titanium alloys. Petroleum fuel has insufficient heat sink for actively cooling the entire airplane, so either hot structure or insulated structure are contenders for Mach 4.5 aircraft. Figure 35 shows an advanced hot structure for wing and fuselage application. The structural panels are formed by two beaded skins providing efficient load support (ref. 19). An advanced insulated structure is shown for the fuselage in figure 15 for an

integral tank. In figure 36, lightweight heat shields are supported by a load bearing metallic insulation. Insulation packages are also attached to the structure at the ring frames. Numerous heat shield and insulation package concepts have been suggested and developed to some degree but further study and tests are needed to identify the one best radiative thermal protection system.

APPENDIX B

STRUCTURAL RESEARCH PACKAGES

The high speed research airplane is intended to provide test results that would form a basis for rational selection of optimum structural and propulsion technologies for future vehicles. Therefore, the various R&D programs may, after analysis and ground-based tests, culminate in flight testing the more promising solutions. To accommodate such tests the payload bay, wings, and entire TPS are replaceable, and various attachments are providing for mounting additional fuel tanks and research engines. As shown in figure 6, the HSRA is fabricated in several detachable sections: a forward fuselage, a payload bay, the integral tank, wings, heat shields, leading edges, nose cone and control surfaces. As described in the HSRA structure section, the above components have built-in field splices or bolted attachments providing relatively simple replacement of components. The technology to be flight tested would be incorporated into the appropriate replaceable part of the HSRA to form a research package. Potential research packages are described in the following subsections for the purpose of illustrating the structural test capability of the research airplane, and as indicated in reference 1, propulsion research can be performed simultaneously.

Liquid Hydrogen Tanks

Liquid hydrogen offers promise as an efficient clean fuel for future aircraft. Presently long life tanks, efficient reusable TPS, and onboard inert purge systems have not been developed. Moreover, lightweight noncircular high volumetric efficiency tanks of either integral or nonintegral construction are a subject of research. Future research packages may include tests of a

nonintegral tank as shown in figure 37. In this research package the tank is suspended from the bulkheads at each end of the payload bay. Hard point attachments are needed in these bulkheads. The aft support struts transmit thrust to the tank, therefore, the aft hard points are aligned with the longerons which are attached to the beam ends of the thrust structure.

Figure 7 illustrates an integral LH_2 tank research package. This tank supports body loads, therefore it is rigidly attached to the basic airplane structure forward and aft of the payload bay. A beaded transition structure is attached to basic airplane bulkheads in a manner identical to the basic airplane payload bay structure. Consequently, no additional hardware is needed for the basic airplane to enable tests of the integral tank research package.

Advanced Primary Structures

Either the payload bay structure or the wings of the basic airplane may be replaced by advanced primary structure research packages. With hydrogen fuel, a large cooling capacity is available, so research of cooled primary structures has been underway to determine the potential of extending low temperature structural materials to high speed aircraft application. Figure 37 illustrates an actively cooled structure research payload. The cooled structure forms the aerodynamic surface or outer mold line; whereas, the basic airplane structure is covered with heat shields. Consequently, the two structures are not inline requiring a transition structure at the ends of the research package payload bay structure. This transition structure, similar to that of the integral tank, is attached to the bulkheads identically to the attachment of the basic payload bay structure.

A second structural research package may be made of advanced panels instead of the Z-stiffened skins of the basic airplane payload bay or wing. Such a structure, shown in figure 38, would be an insulated structure; however, the baseline heat sink heat shields are suitable for such structural testing. The advanced tubular panel research structure would be attached to end bulkheads identically to the baseline payload bay structure.

A third structural research package may be an advanced hot structure, as shown in figure 35. As with the actively cooled structure, the hot structure forms the mold line requiring end transition structures to the baseline structure. These transition structures attach to the baseline structure bulkheads identically to that of the baseline payload bay structure.

Each of the advanced actively cooled, insulated and hot structures may also form research packages as wing structures. The wing body joint of the basic airplane has been designed to allow attachment of various wing structures to the body using the same body attachments.

Advanced TPS

The entire TPS is replaceable in the baseline HSRA design. Moreover, space between the heat shields and the primary structure has been provided in the baseline design to accommodate research package thermal protection systems.

Radiative metallic heat shields and packaged high temperature insulation, as shown in figure 36, may be tested on any part or all of the basic airplane surface. Flight testing can determine the effectiveness of the shield support and attachment for resistance to flutter and leakage of hot boundary layer air into the insulation space. Venting and sonic fatigue effects are other objectives of flight testing. With the basic airplane, the shield standoffs

are removed and replaced by circumferential and spanwise strips of metallic load bearing insulation on the body and wing, respectively. The heat shields and the insulation packages are designed to support venting pressure differences. The inner surface of the package may have phase change material (PCM) attached by enclosing PCM in a tubesheet facing on the insulation package. This facing is perforated to allow pressure equilization of the insulation voids and the space between the insulation and structure.

A second advanced TPS is actively cooled heat shields. For areas of high heat flux, such as the lower aft fuselage with scramjet propulsion, active cooling of superalloy shields may provide long-life, fail-safe TPS. Coolant distribution can be located in the space provided between the shields and primary structure.

A third advanced TPS may consist of a superalloy shell suspended at Fuselage ends. This one-piece heat shield avoids high temperature seals and slip joints of radiative shields and minimizes attachments to the structure. As a research package, this TPS may be attached to the baseline structure at the ends of the payload bay wherein one end of the shell slides. Insulation packages would be attached to the primary structure. An attachment ring may be required to support the heat shield shell, but this would be added when the research package is installed.

Leading Edges

The basic airplane leading edges are replaceable permitting research of advanced leading edge concepts. Radiative systems include coated refractory metal and carbon composites. Long life may require actively cooled leading edges for the hypersonic applications. The need for hydrogen fuel at hypersonic speeds provides a large coolant heat sink which will be

required for engines and probably beneficial for the leading edges and nose cap. Secondary coolant flow within plate-fin-type sandwich leading edge segments of nickel alloy forms a practical approach. The technology involved is similar to that developed for the Hypersonic Research Engine, ref. 20.

Research Engines

The lower body of the HSRA is contoured to suit testing scramjet engines, and the lower wing surface is flat to suit testing ramjets and turbojets. Moreover, bulkheads are provided within the body tanks to enable various quantities of different fuels for the basic rocket propulsion of the HSRA and for JP-fueled research engines. The heat shields may be locally removed to enable attachment of engines to the airframe, and the heat shield mold line contour may be modified by different shields of the basic design to enable inlet and exhaust research for a scramjet. Hard points are provided in the wing (fig. 39) and body structures to enable future testing of research engines without modification of the airframe. As indicated in figure 39, a three-point suspension is provided for each engine. One attachment is stationary except for rotation, the other two attachments are slip jointed to permit differential thermal expansion between the wing structure and engine, while simultaneously sustaining all engine forces without relative movement except that induced by thermal expansion.

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Table I.- Structural mass estimates for the high speed research airplane.

Component	Initial Estimate				Final Estimate			
	Kg	(lb_m)	Kg/m^2	(lb_m/ft^2)	Kg	(lb_m)	Kg/m^2	(lb/ft^2)
Wing	421	(929)	5.30	(3.42)	650	(1432)	8.16	(5.26)
Tail	112	(247)	4.73	(3.05)	166	(366)	9.31	(6.00)
Surface controls*	317	(700)	13.6	(8.75)	99	(219	4.25	(2.74)
Body	2133	(4703)	6.48	(4.18)	2142	(4722)	6.53	(4.21)
Landing gear	374	(825)	0.70	(0.45)	317	(700)	0.61	(0.39)
Structure total	3358	(7404)	6.21	(4.00)	3374	(7439)	6.22	(4.01)
Thermal protection**	3170	(7000)	5.86	(3.78)	3170	(7000)	5.86	(3.78)
Total	.6534	(14404)	12.1	(7.78)	6550	(14439)	12.1	(7.79)

*Initial estimate is based on structure covered with heat sink shields, which are listed with thermal protection mass; final estimate is based on structure that has heat sink skins and the skin mass is listed with thermal protection mass.

**Initial and final estimates are each based on the same analysis.

Table II. HSRA performance with Lockalloy and beryllium heat sink heat shields.

HSRA performance conditions		ΔT = 811°K (1000°F) with insulation 680 kg (1500 lb)		ΔT = 533°K (500°F) Without insulation		
		Beryllium		Lockalloy	Beryllium	
Gross weight kg(lb_m)	M = 10 q = 23.9 kN/m^2 (500 psf)	3175 (7000)	2722 (6000)	3175 (7000)	2722 (6000)	3175 (7000)
		>45360 (100,000)	42185(93000)	37649(83400)	35834 (79000)	38102 (84200)
Cruise time, min. or (max. Mach number)	M = 10 q = 23.9 kN/m^2 (500 psf)	6.2	50	1.4	1.5	2.1
	M = 10 q = 71.8 kN/m^2 (1500 psf)	2.0	1.3	0 (M = 9.3)	0 (M = 9.3)	0 (M = 9.8)

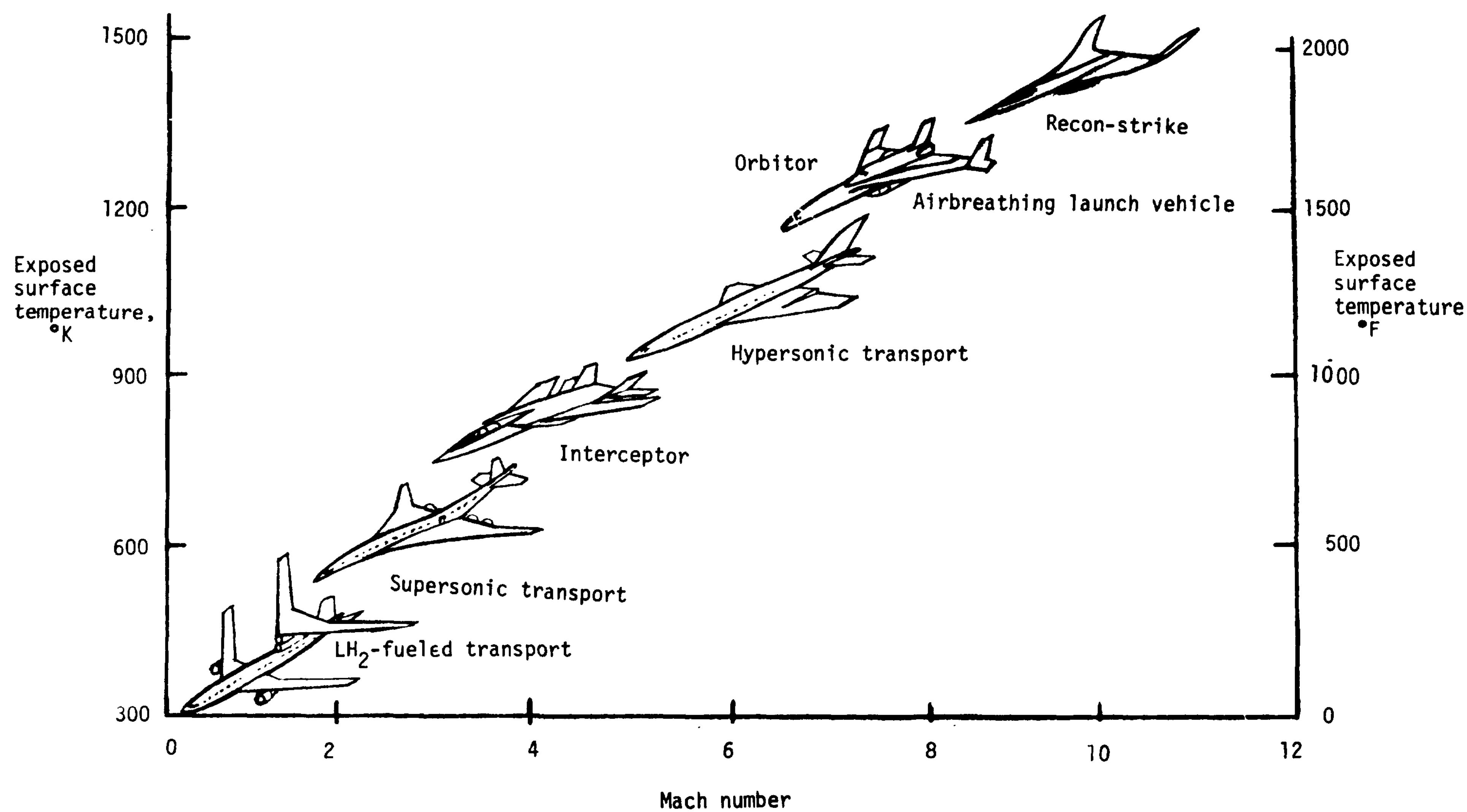


Figure 1.- Exposed surface temperature as a function of Mach number for possible future aerospace vehicles.

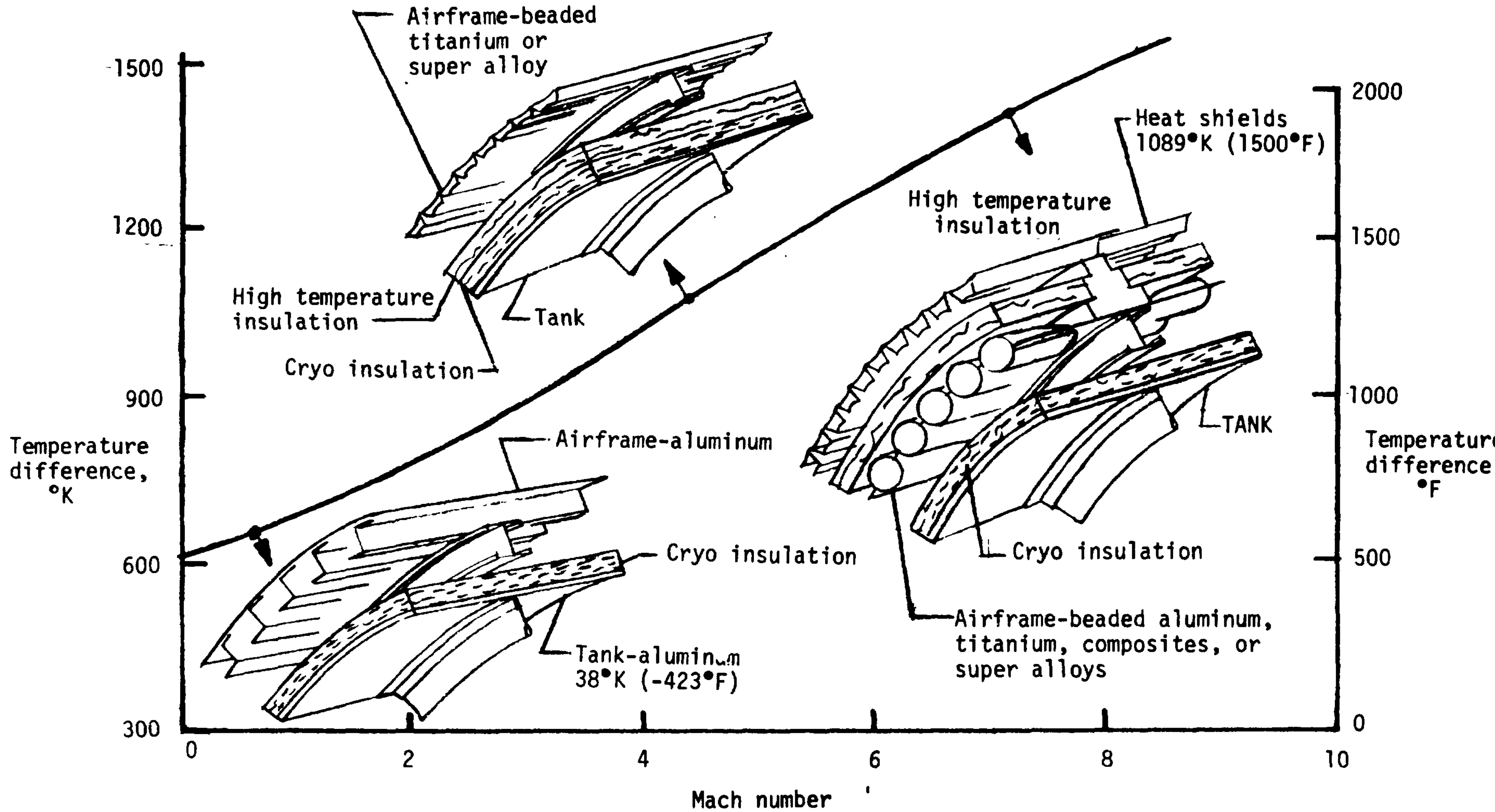


Figure 2.- Temperature difference through wall as a function of Mach number for structural concepts suited to increasing Mach number vehicles.

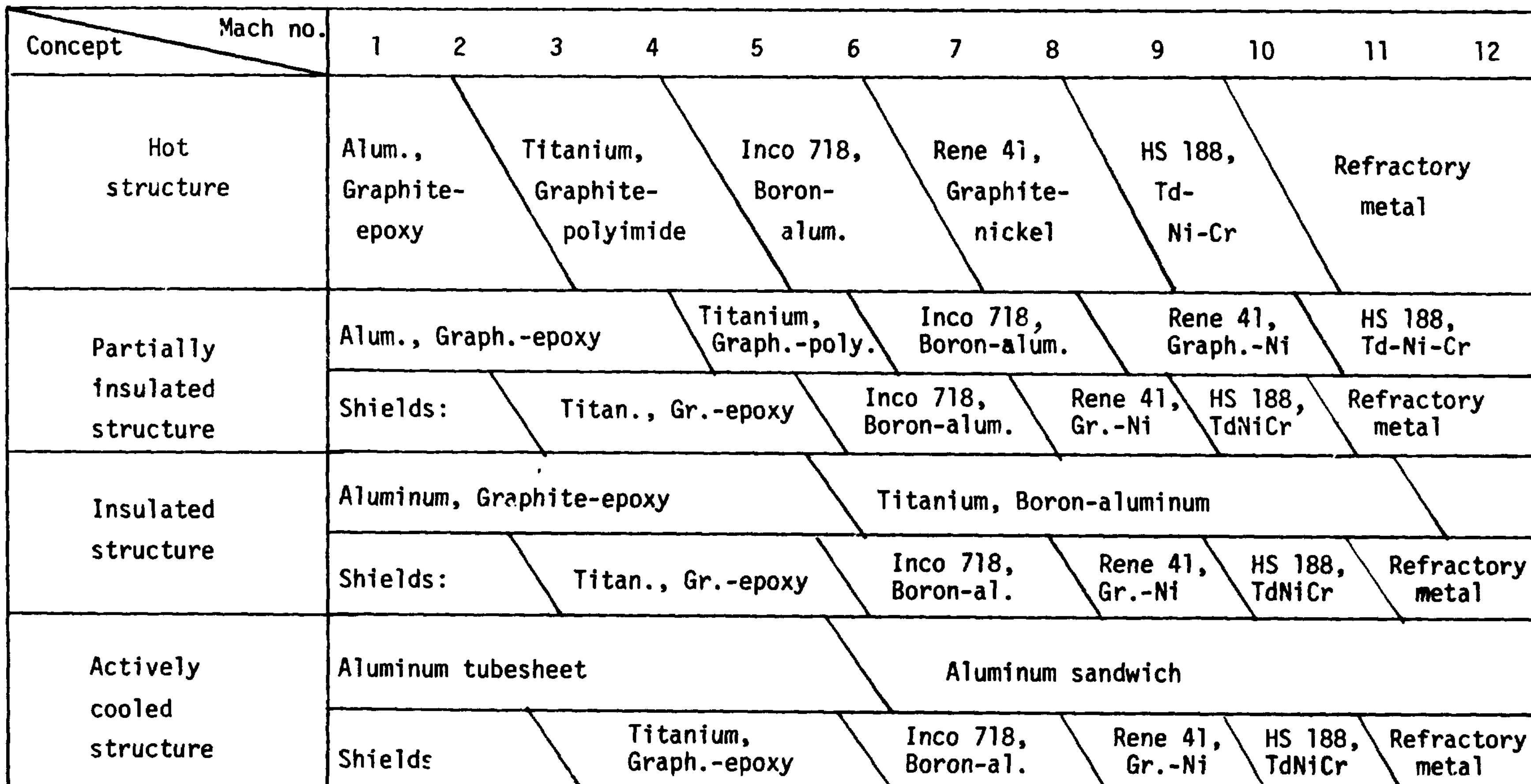


Figure 3.- Effect of Mach number on materials for various structural concepts.

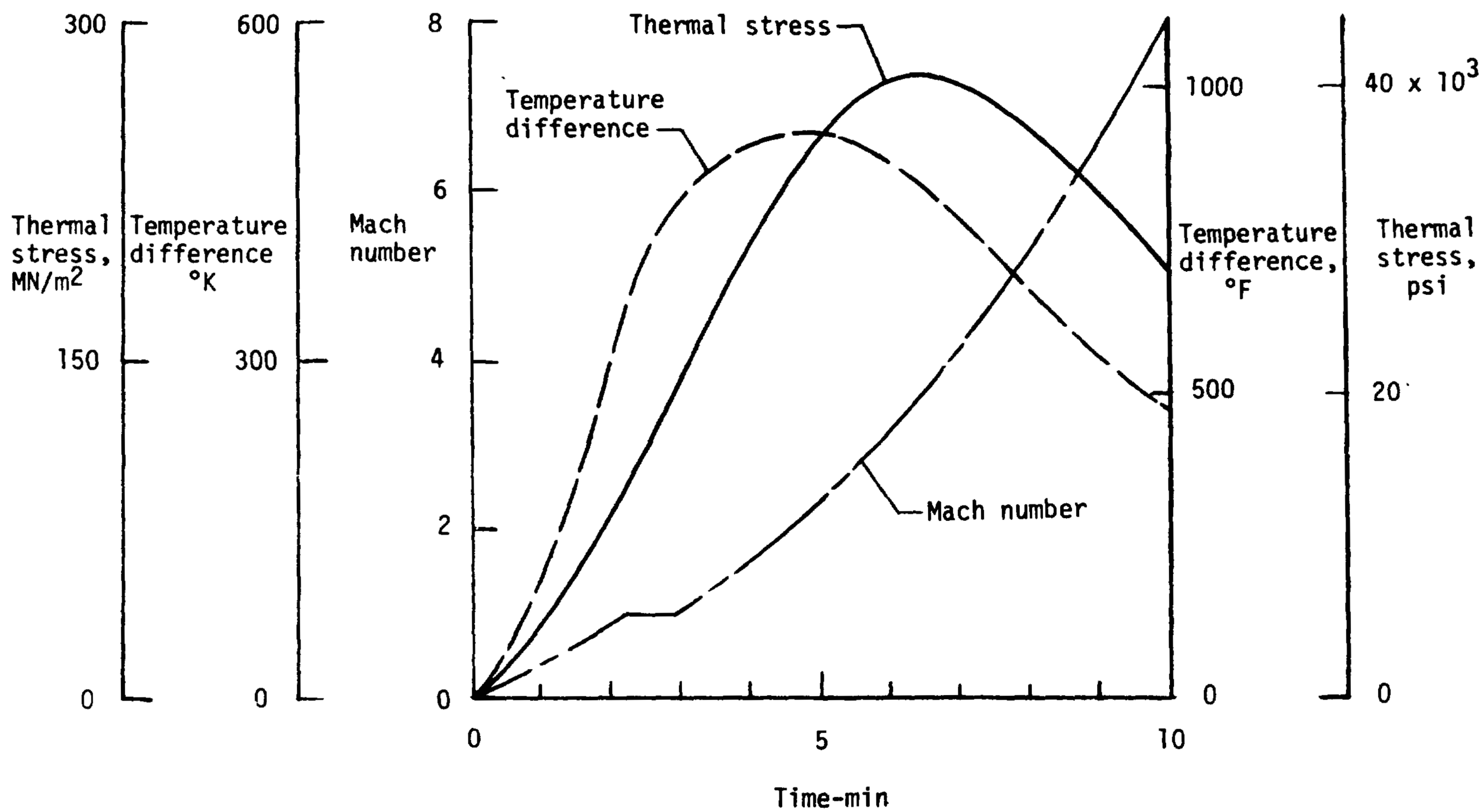


Figure 4.- Thermal stress in simulated flight of hot structure.

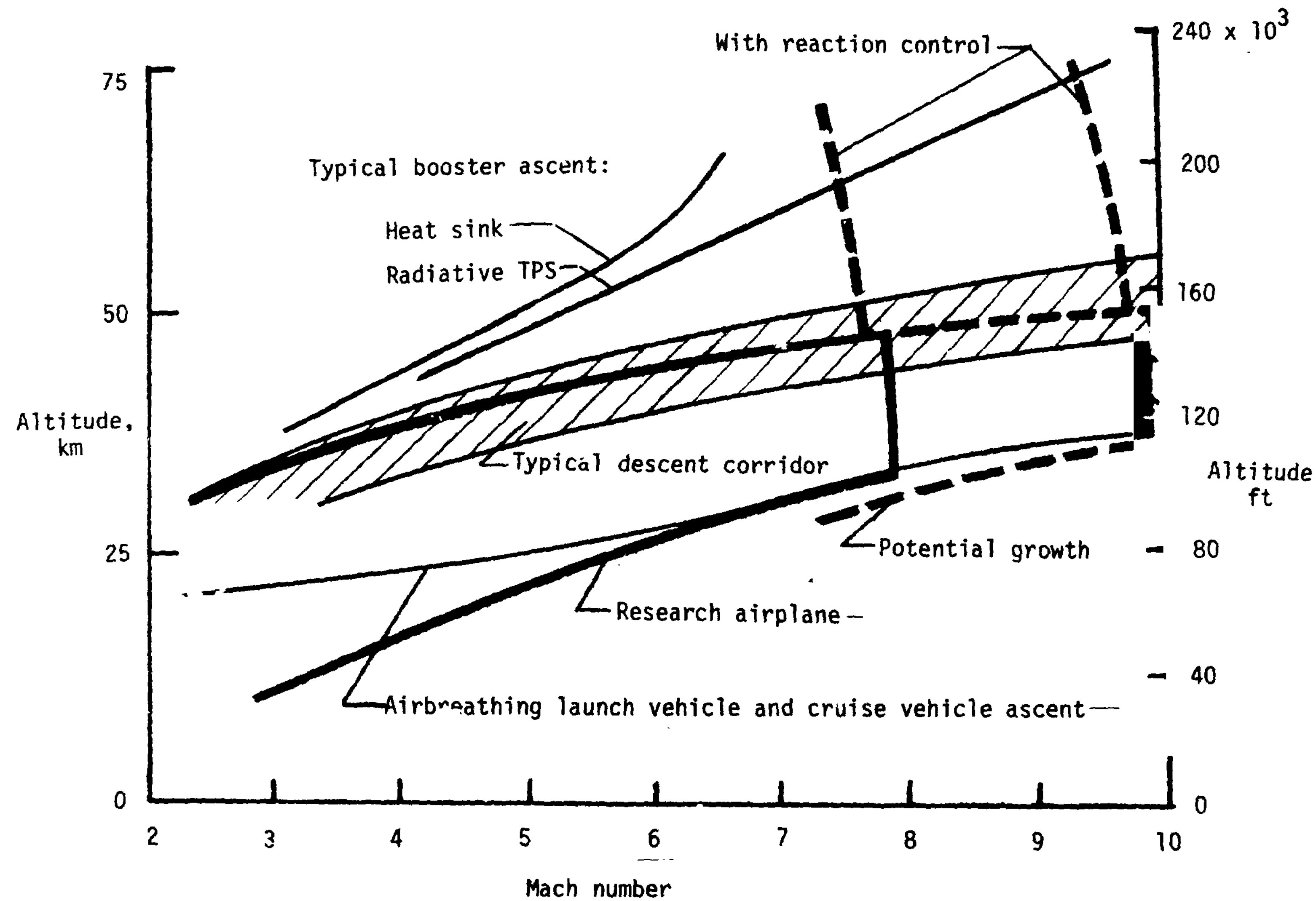


Figure 5.- Flight envelope of high speed research airplane.

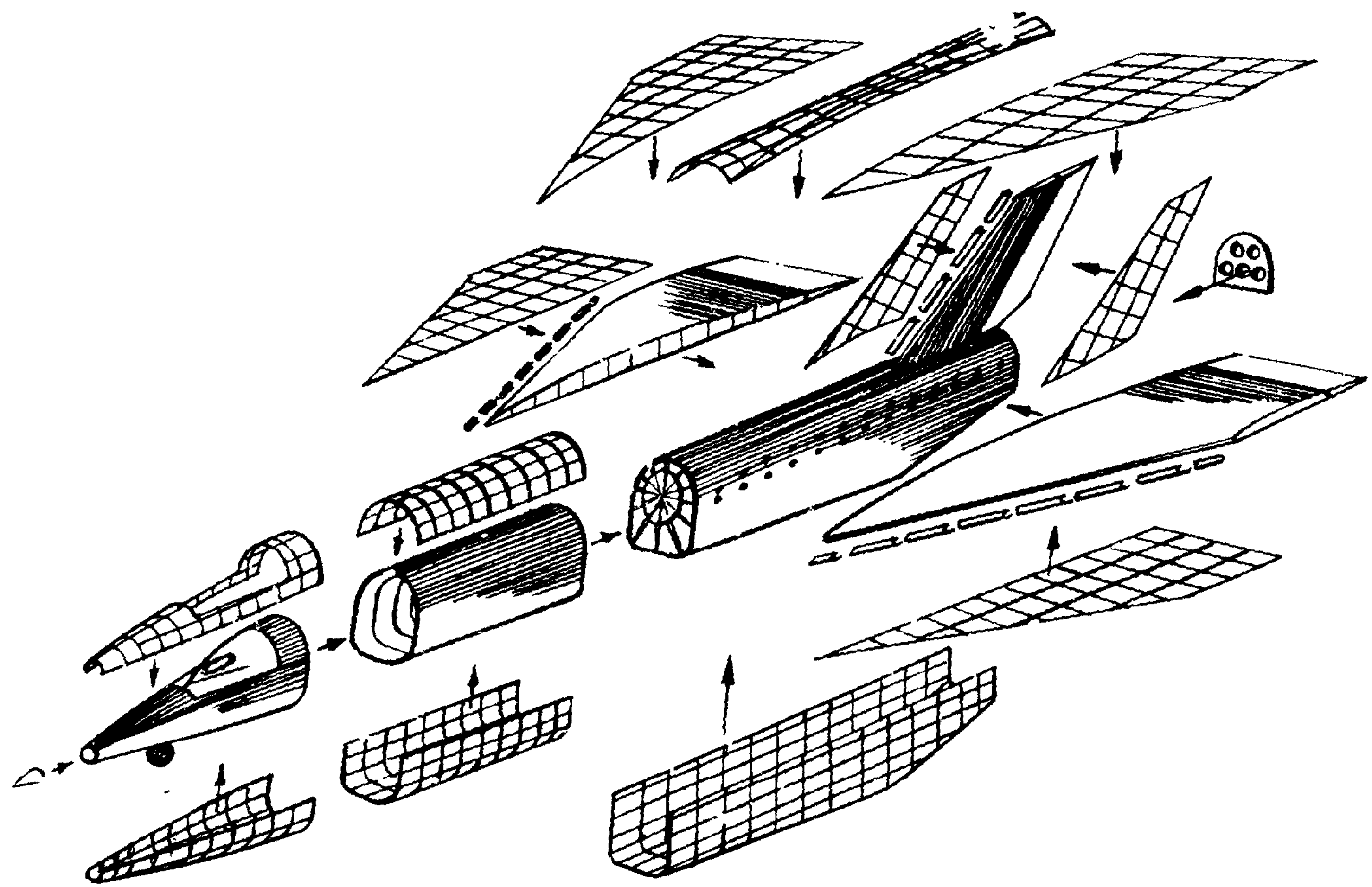


Figure 6.- Exploded view of HSRA showing replaceable sections.

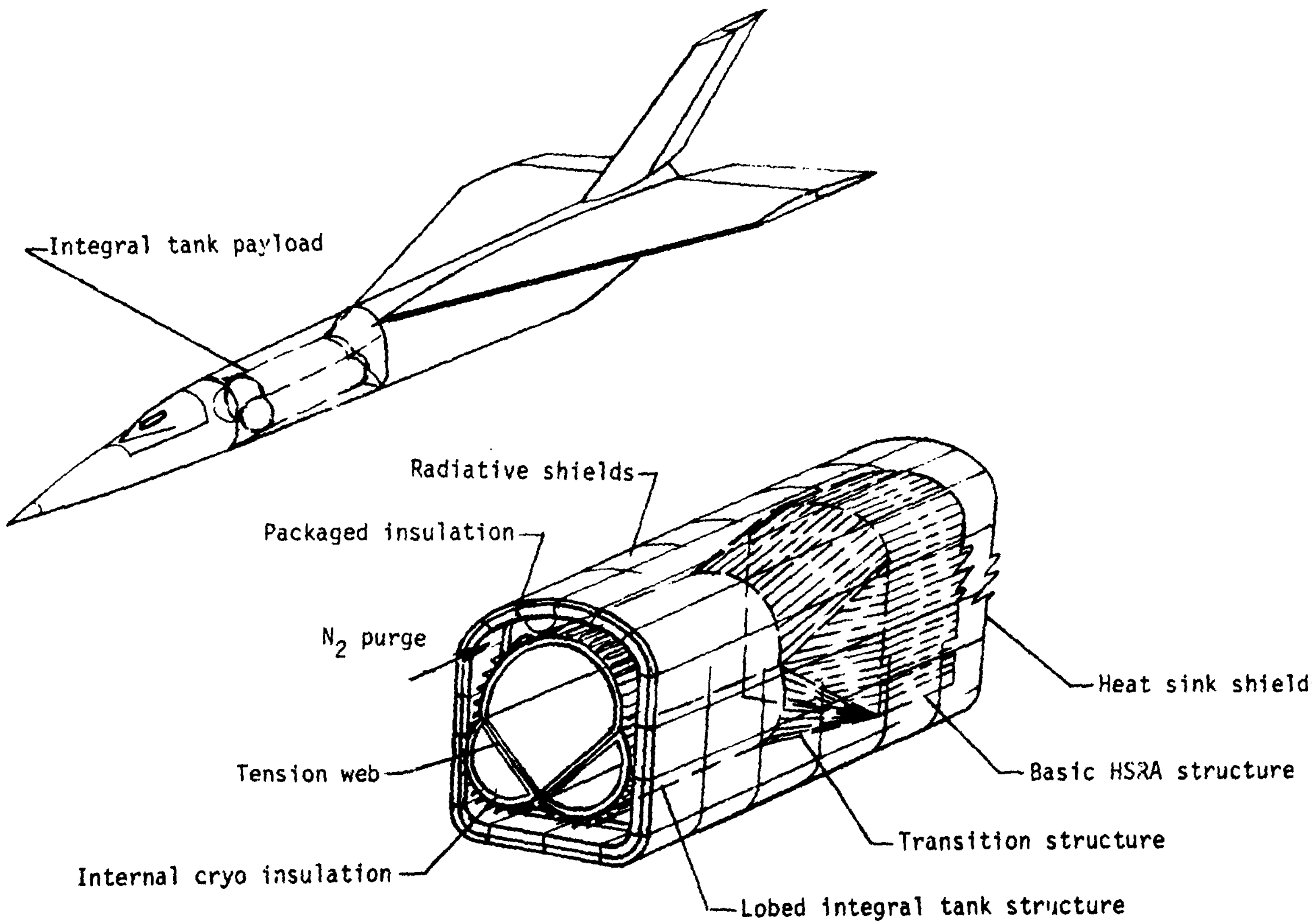


Figure 7.- Integral LH_2 tank research payload in HSRA.

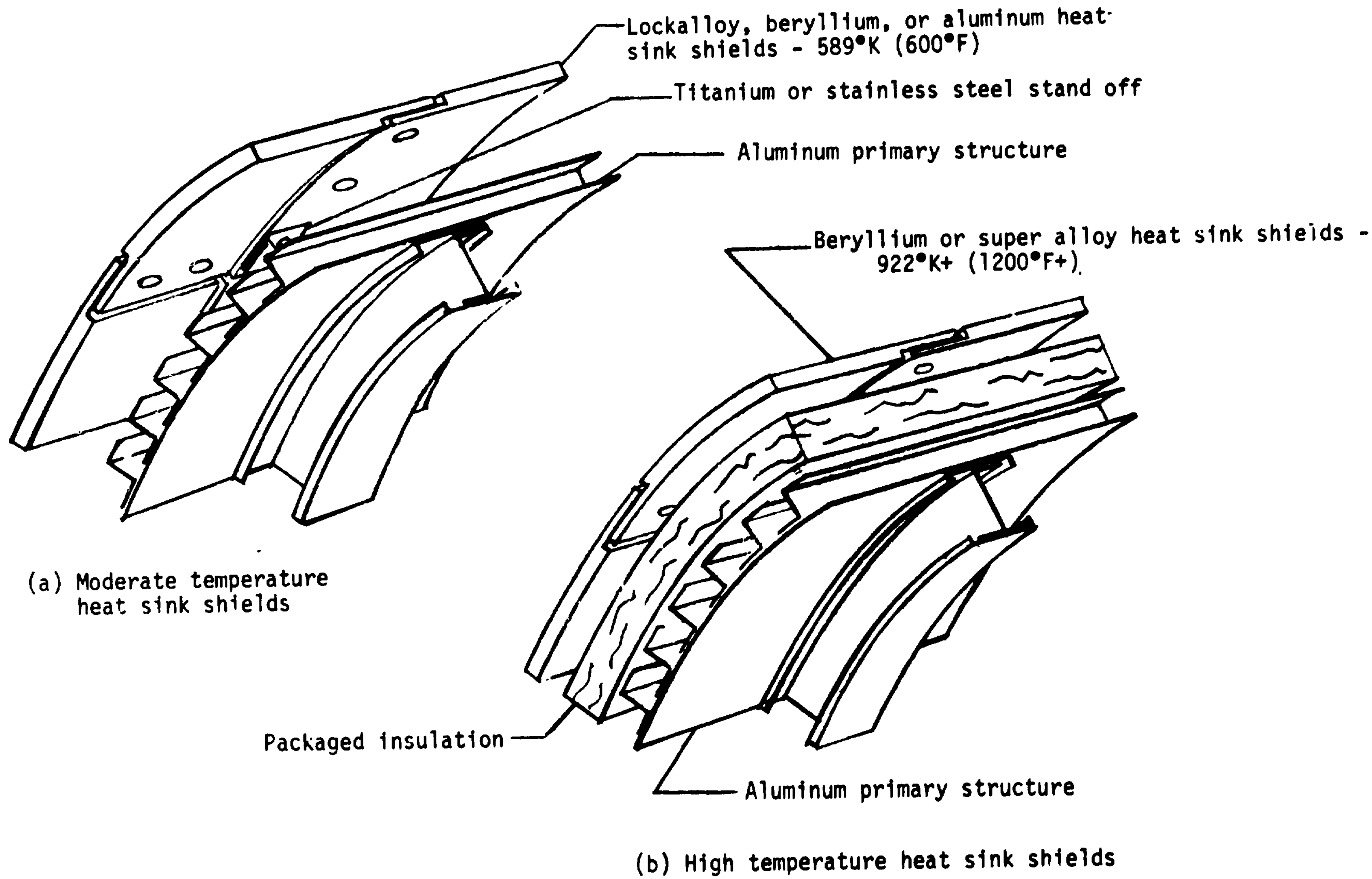
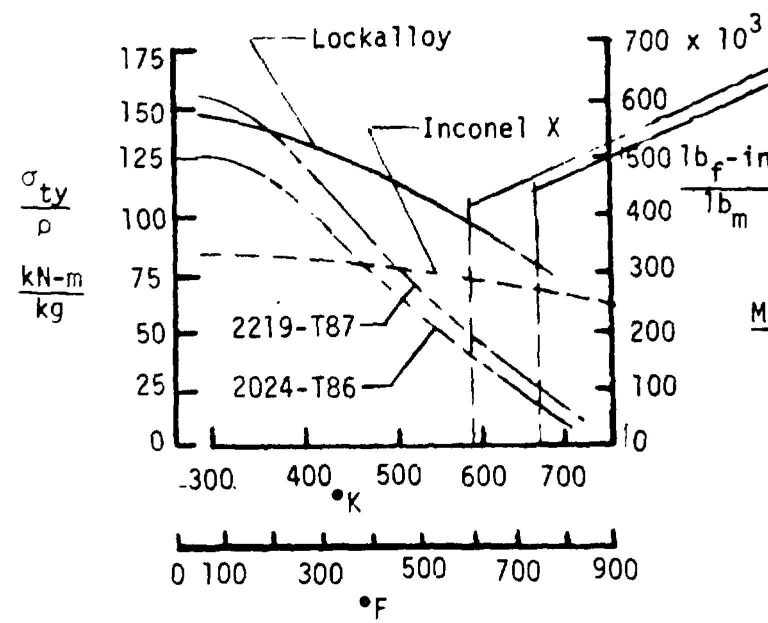
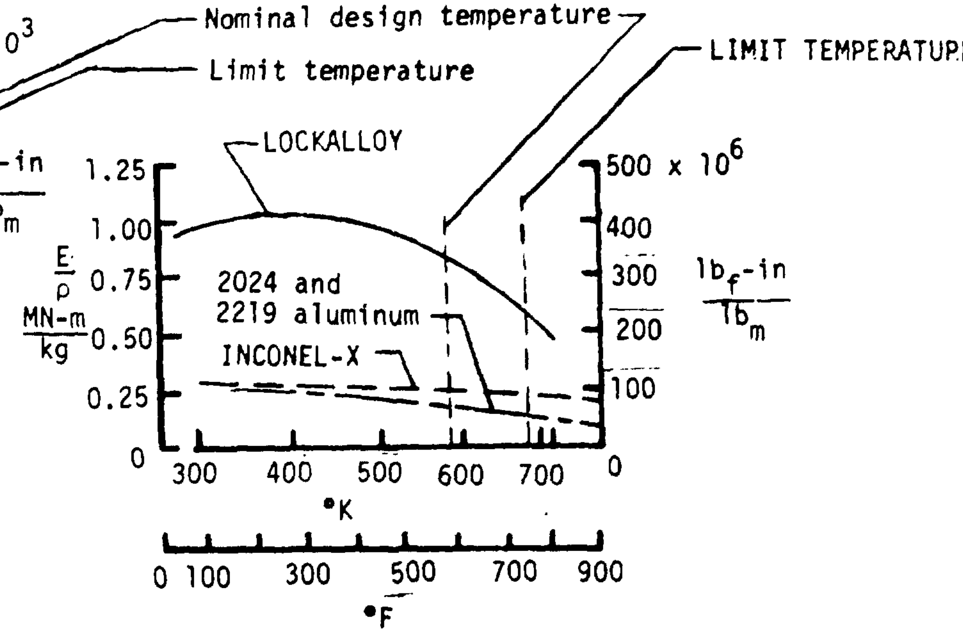


Figure 8.- Heat sink shield alternatives.



(a) Strength-to-density ratio.



(b) Stiffness-to-density ratio.

(c) Properties	Lockalloy (Be-38Al)	Aluminum	Inconel X
K-thermal conductivity	208 W/m-°K (1440 BTU-in./hr-ft^2-°F)	128 (889)	29.2 (202)
C_p - specific heat	1883 J/kg-°K (0.45 BTU/lbm-°F)	962 (0.23)	502 (0.12)
ρ - density	2076 kg/m^3 (0.075 lbm/in^3)	2824 (0.102)	8305 (0.30)
e - ultimate elongation	8%	5%	20%

Figure 9.- Thermophysical properties of various heat sink materials.

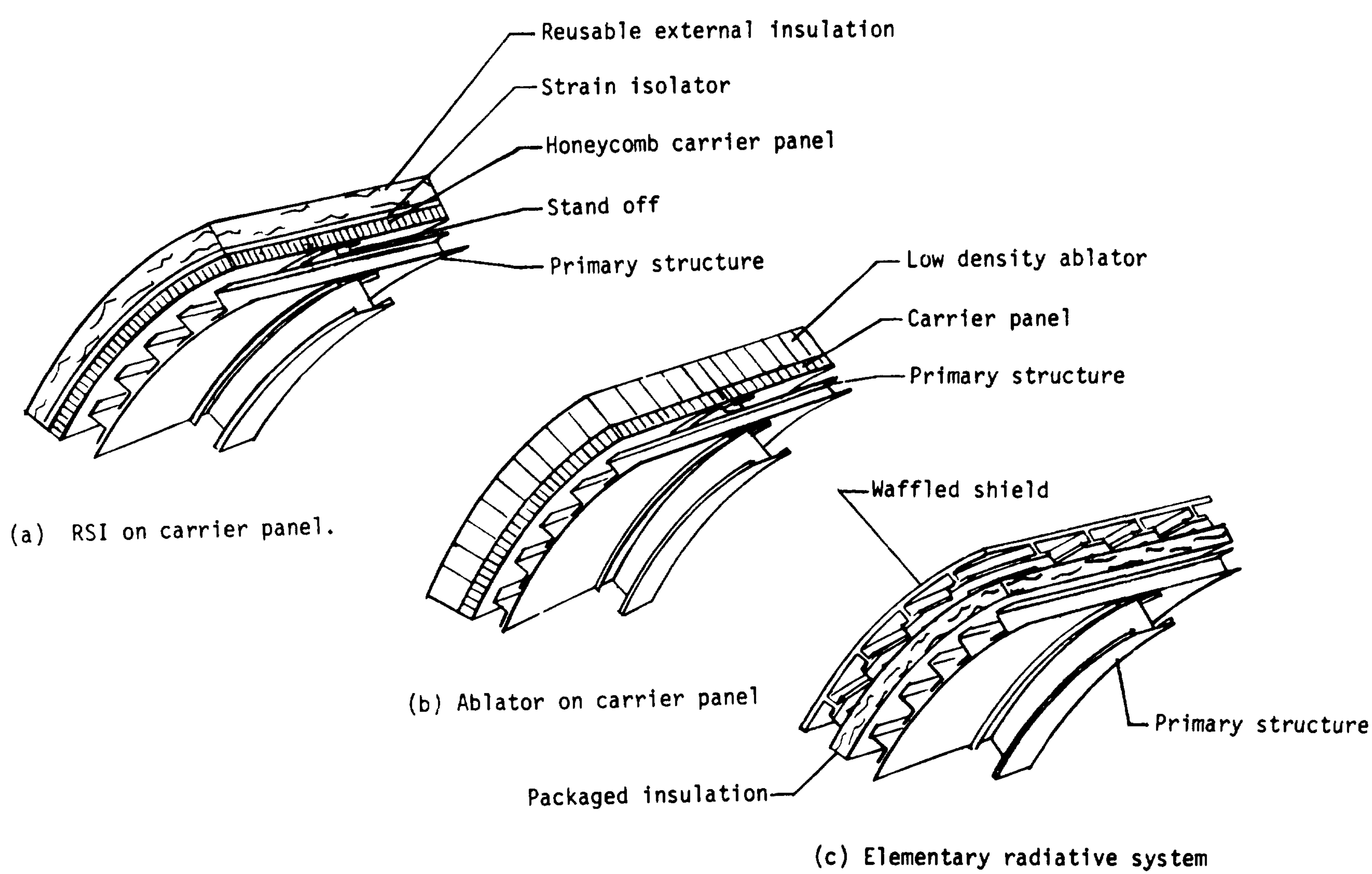
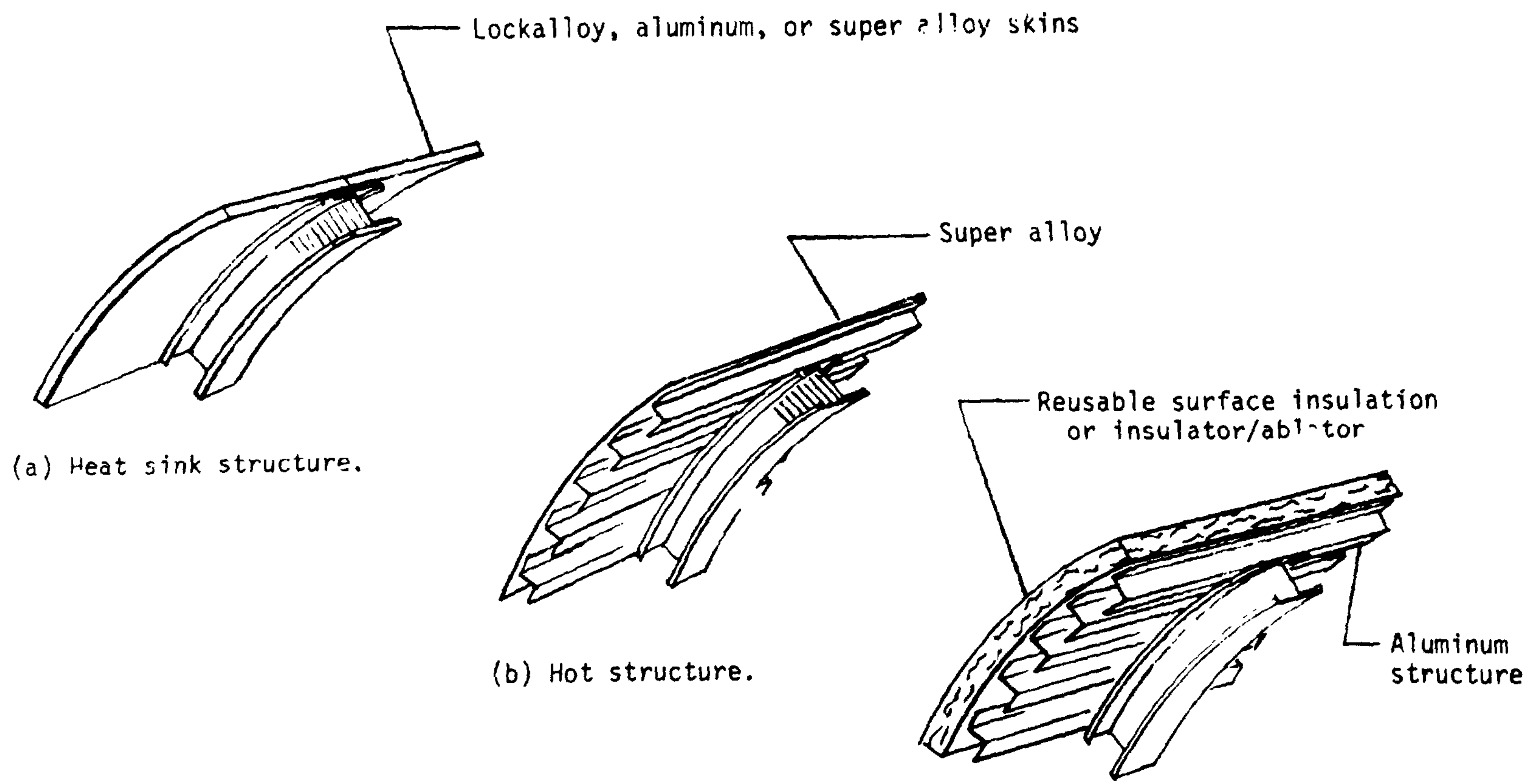


Figure 10.- Alternative thermal protection systems for the basic HSRA.

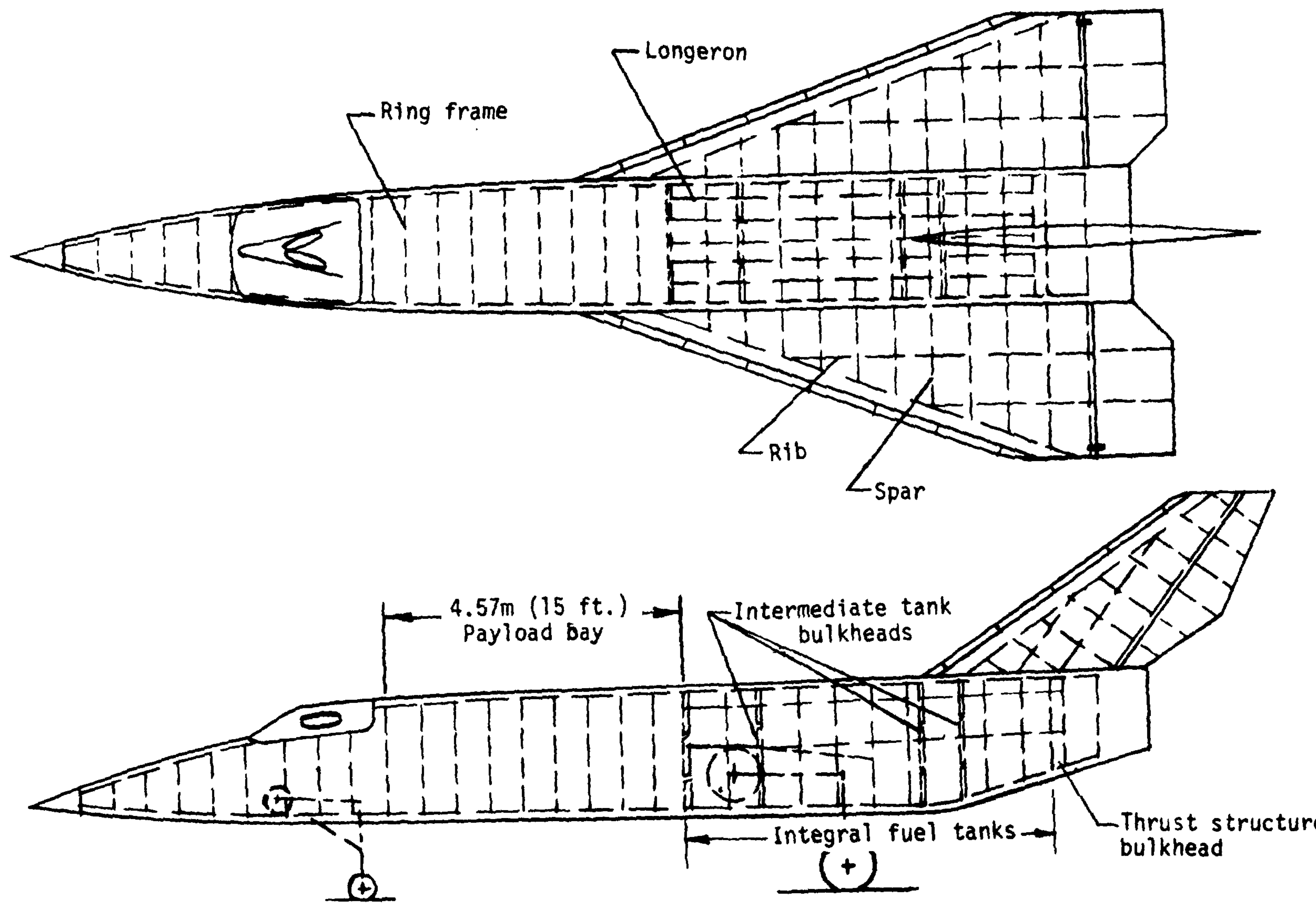


(a) Heat sink structure.

(b) Hot structure.

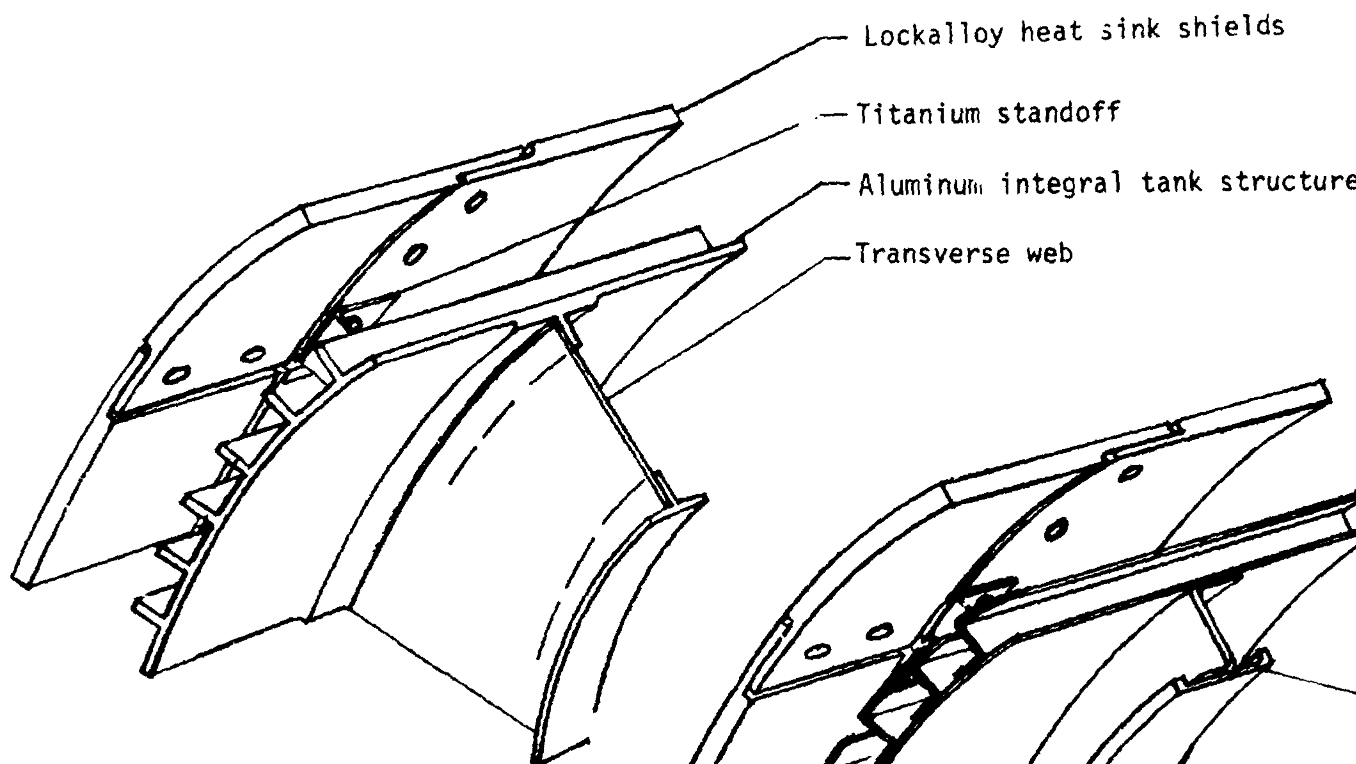
(c) Direct-bond insulators on aluminum structure.

Figure 11.- Alternative structures and TPS for the basic HSRA.

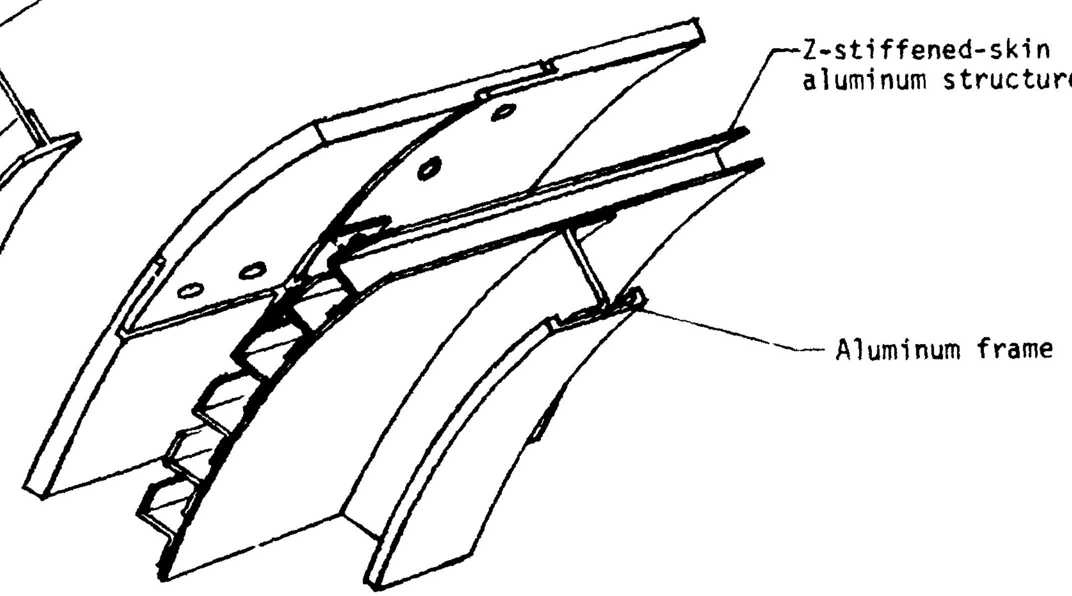


(a) General arrangement.

Figure 12.- High speed research airplane structure.

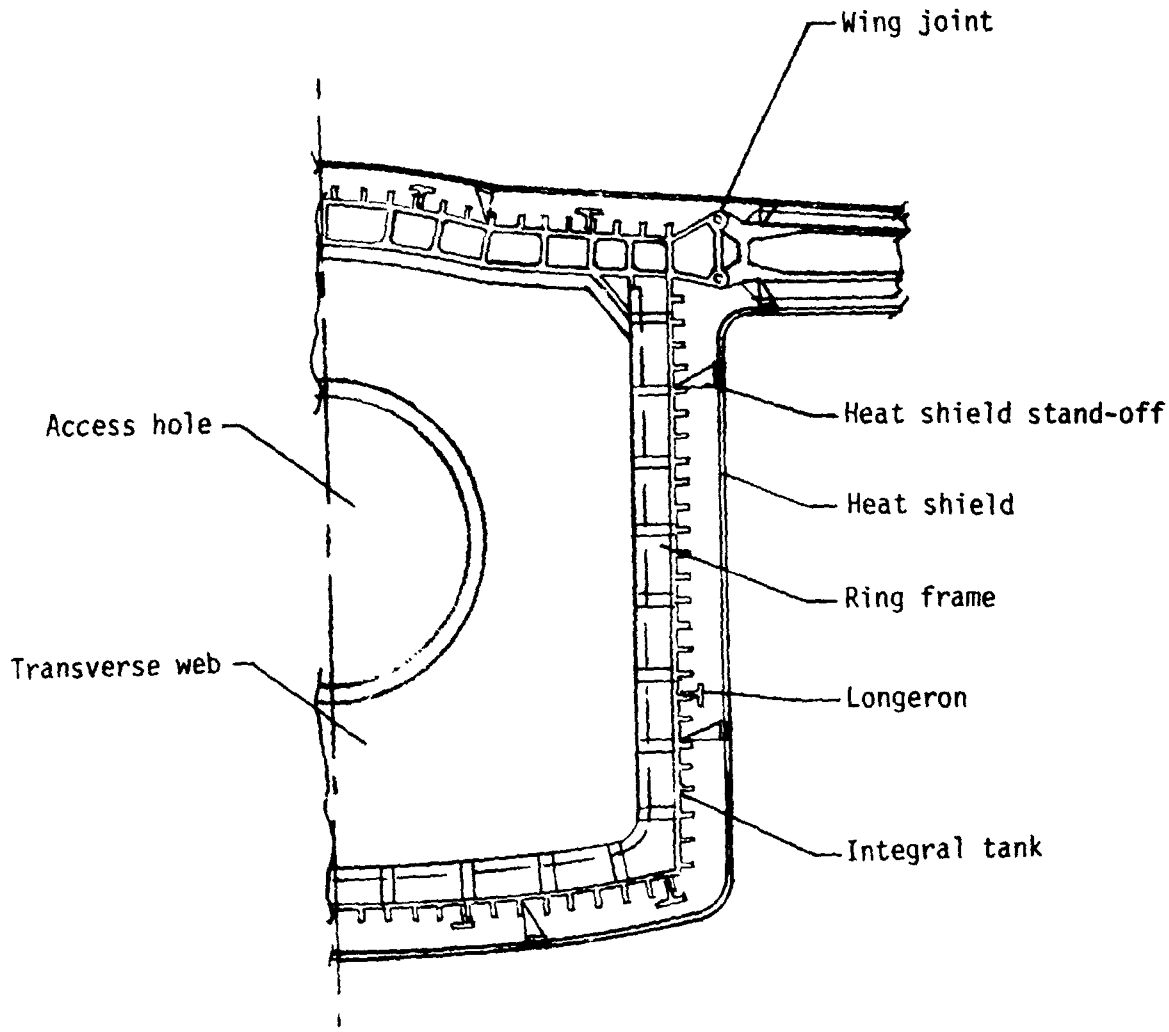


(b) Integral tank wall.



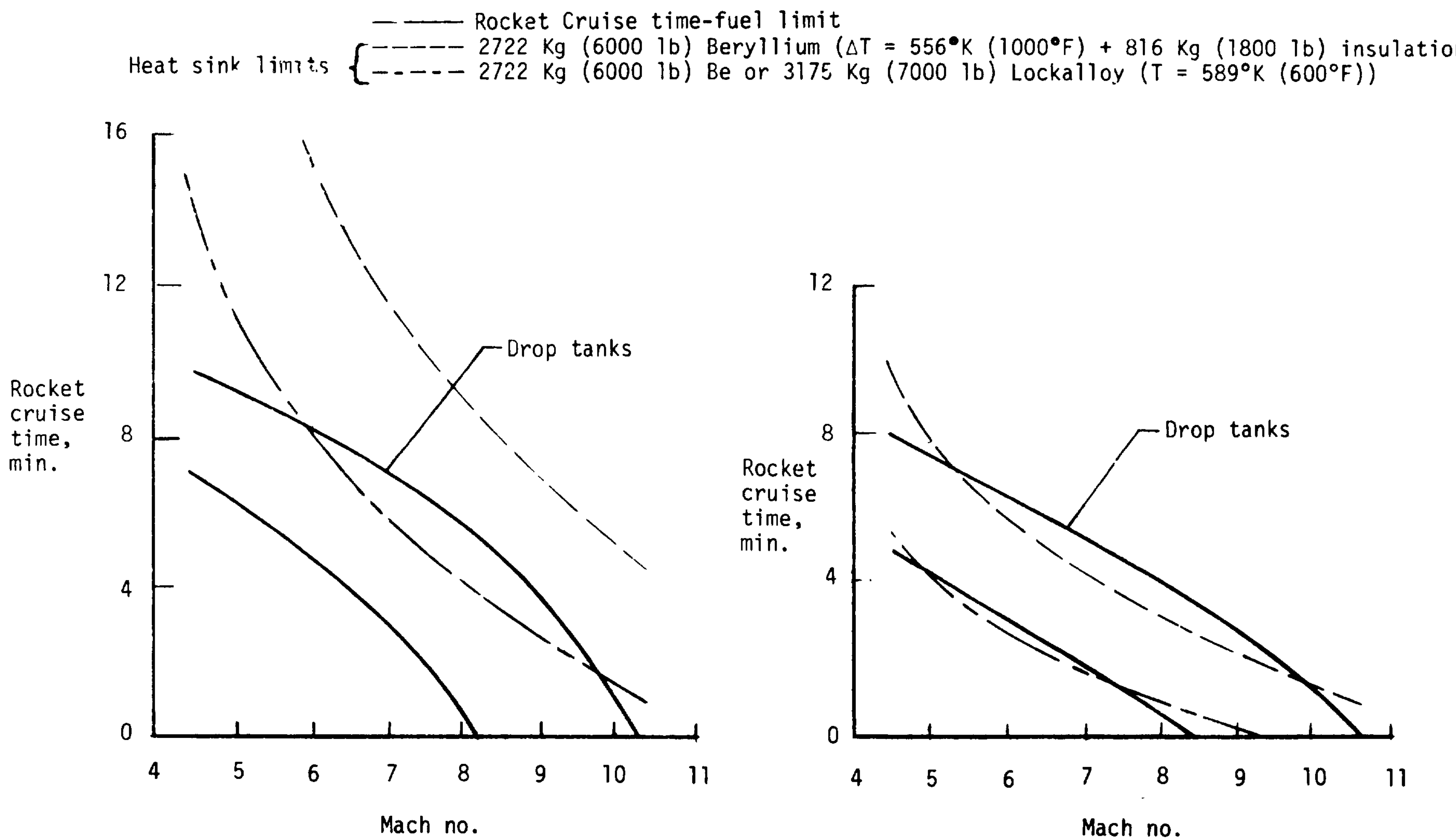
(c) General fuselage wall.

Figure 12.- High-speed research airplane structure.



(d) Section o: integral t.nk and wing joint.

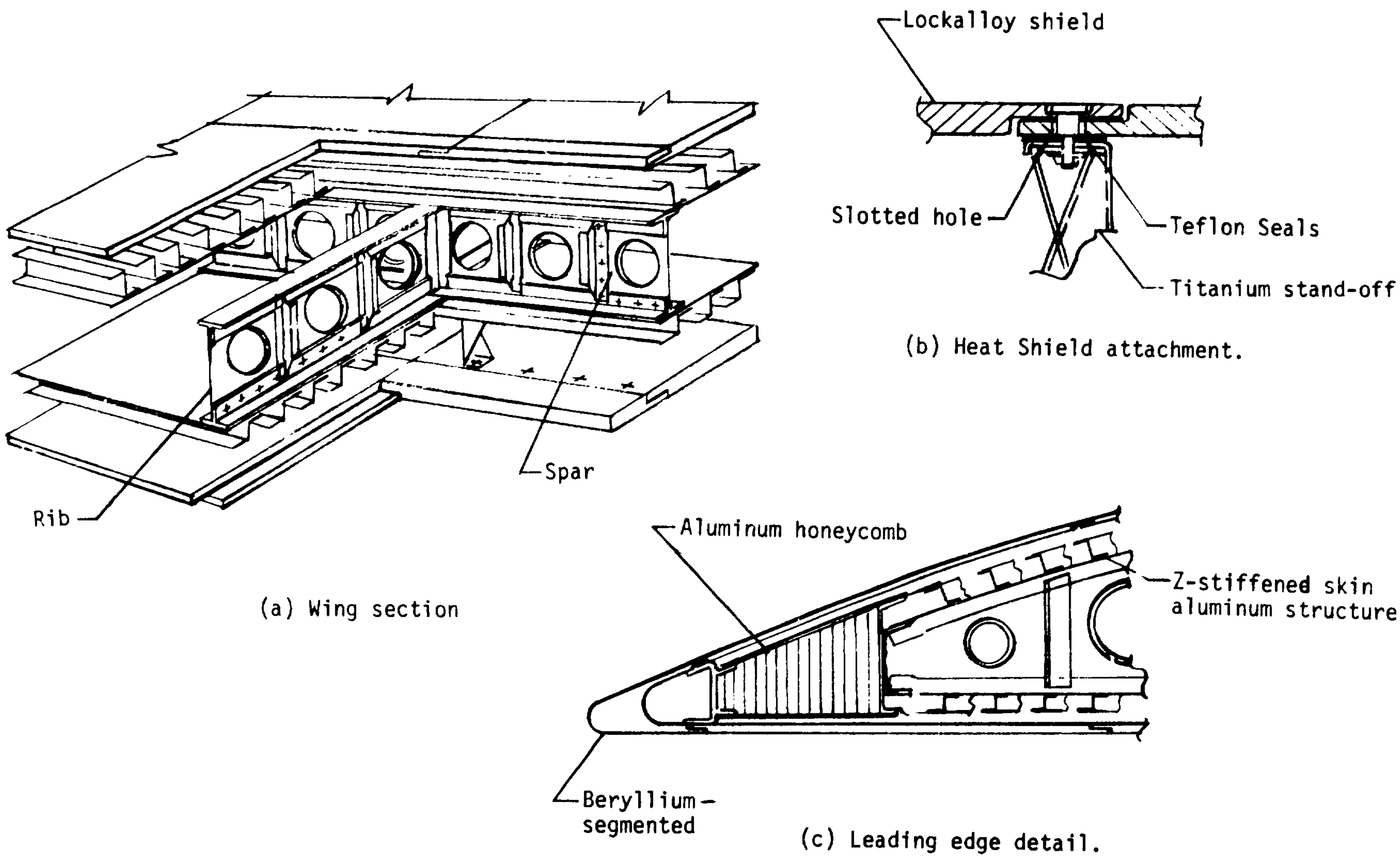
Figure 12.- Concluded.



(a) Dynamic pressure = 23.9 KN/m^2 (500 psf)

(b) Dynamic pressure = 71.8 KN/m^2 (1500 psf)

Figure 13.- Baseline HSRA performance - rocket cruise time.



(a) Wing section

(b) Heat Shield attachment.

(c) Leading edge detail.

Figure 14.- Basic HSRA wing structure.

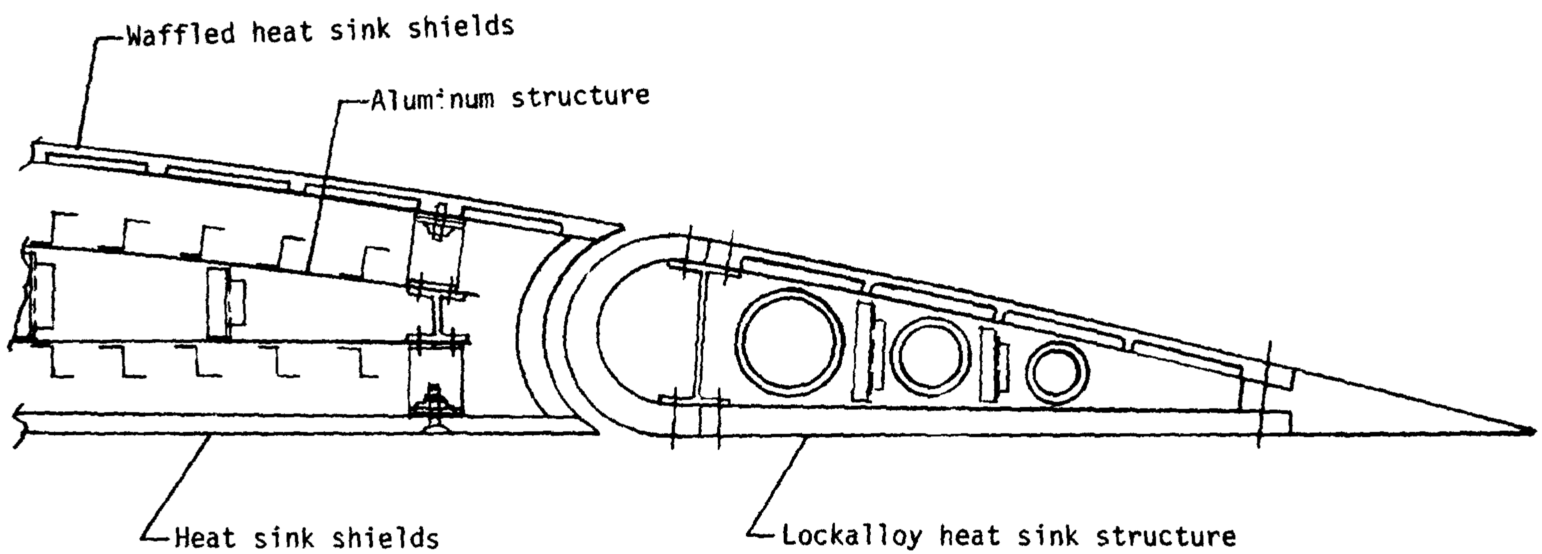


Figure 15.- Basic HSRA control surface heat sink structure.

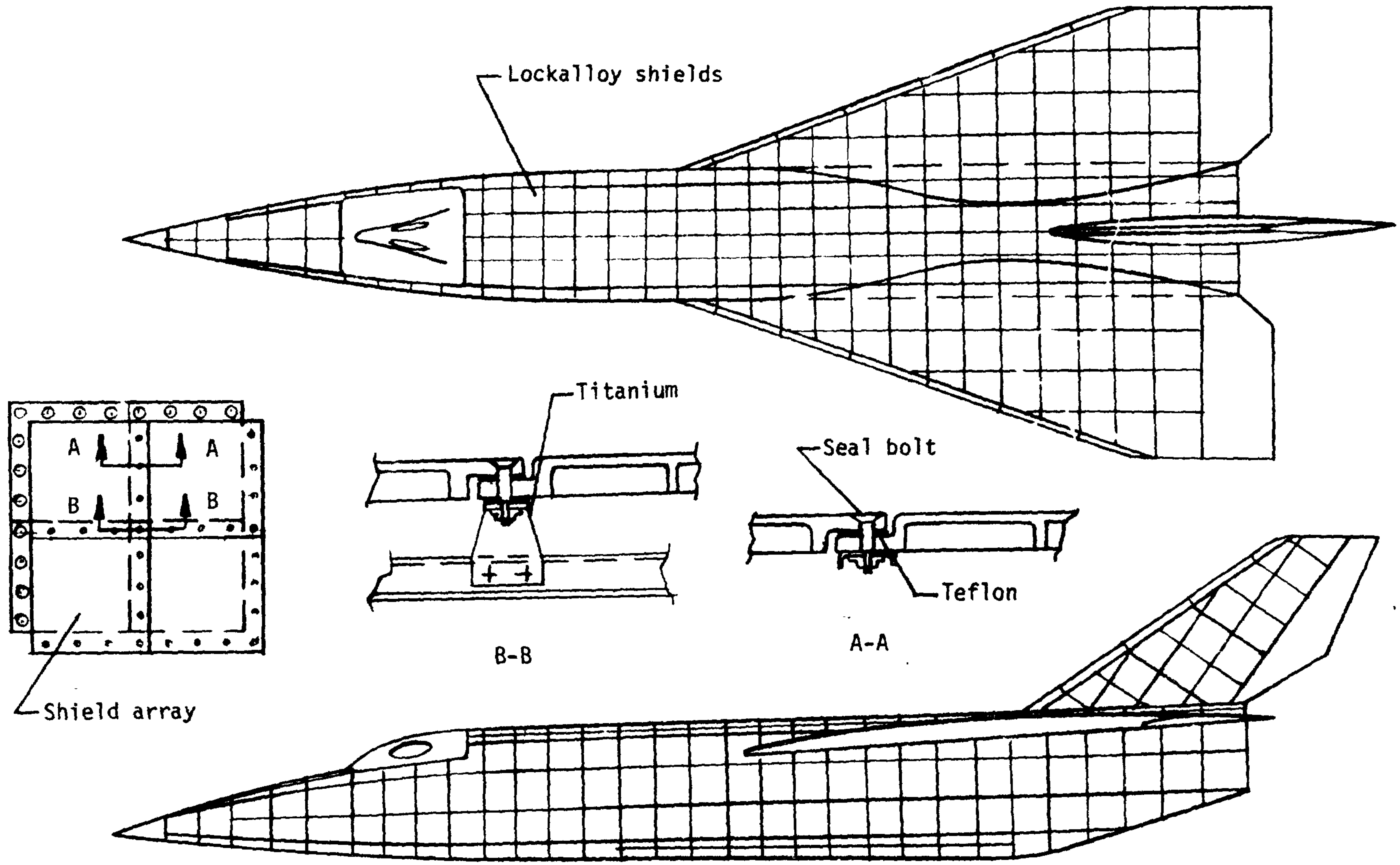


Figure 16.- Basic HSRA thermal protection system

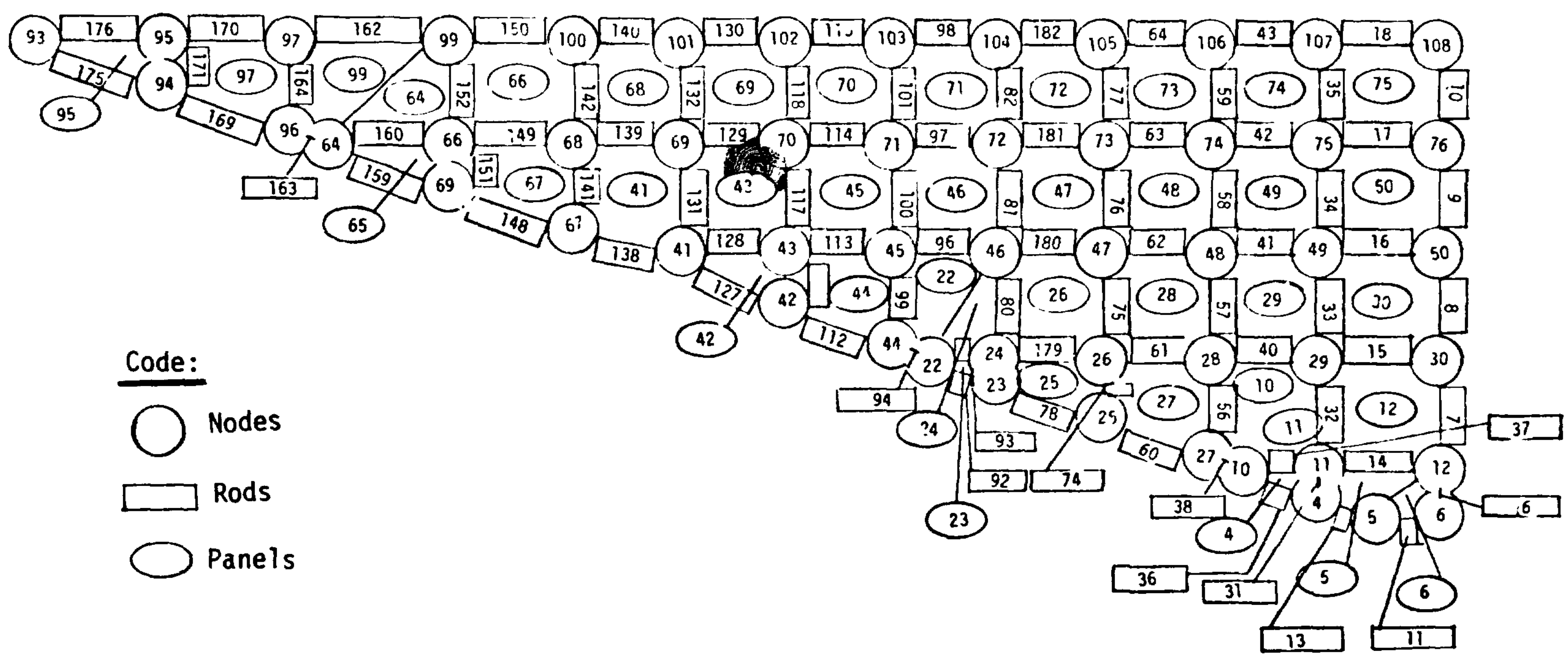


Figure 17.- NASTRAN model of lower surface of HSRA wing.

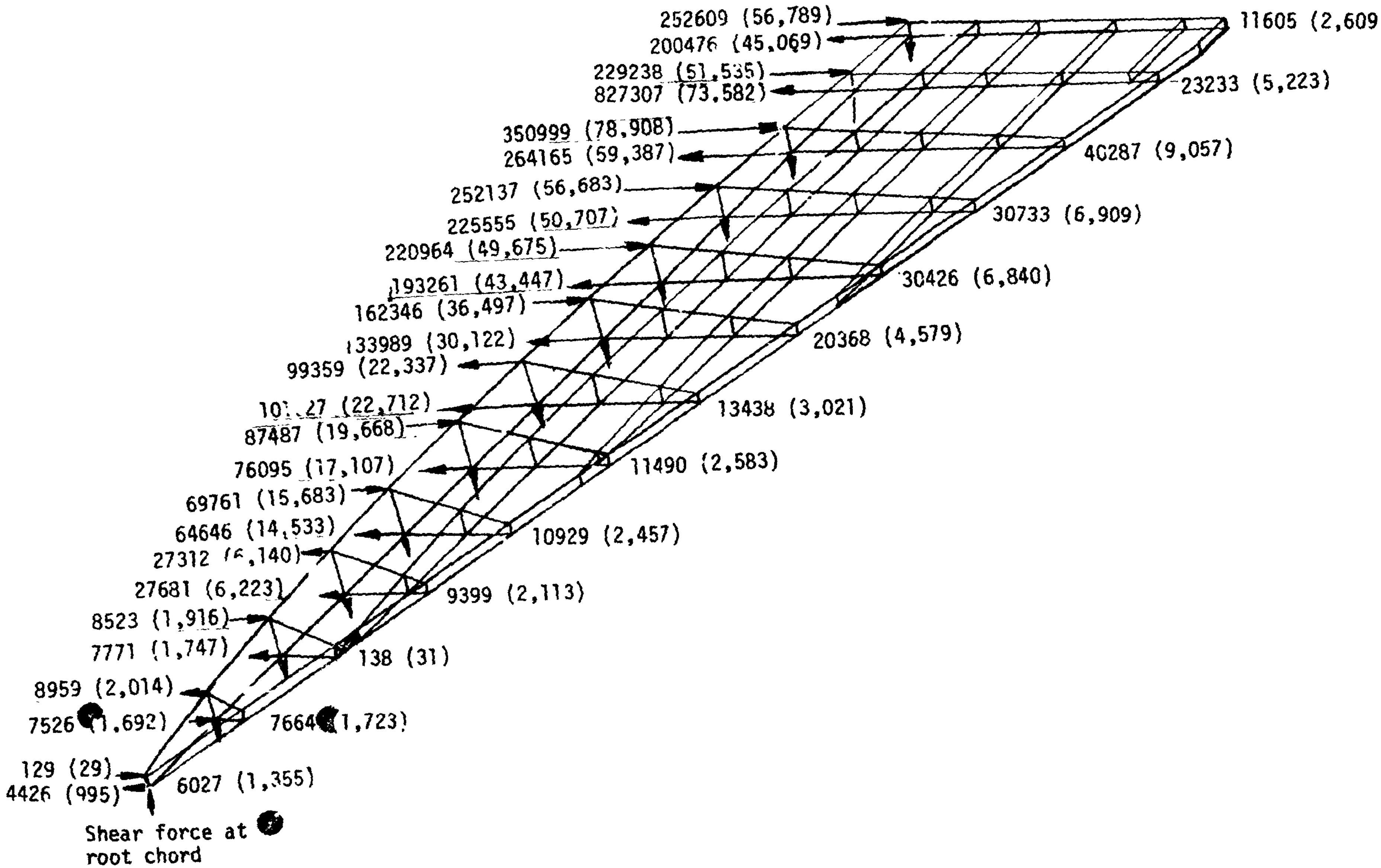


Figure 18.- NASTRAN derived wing reactions for limit load (units are in N and (lb_f)).

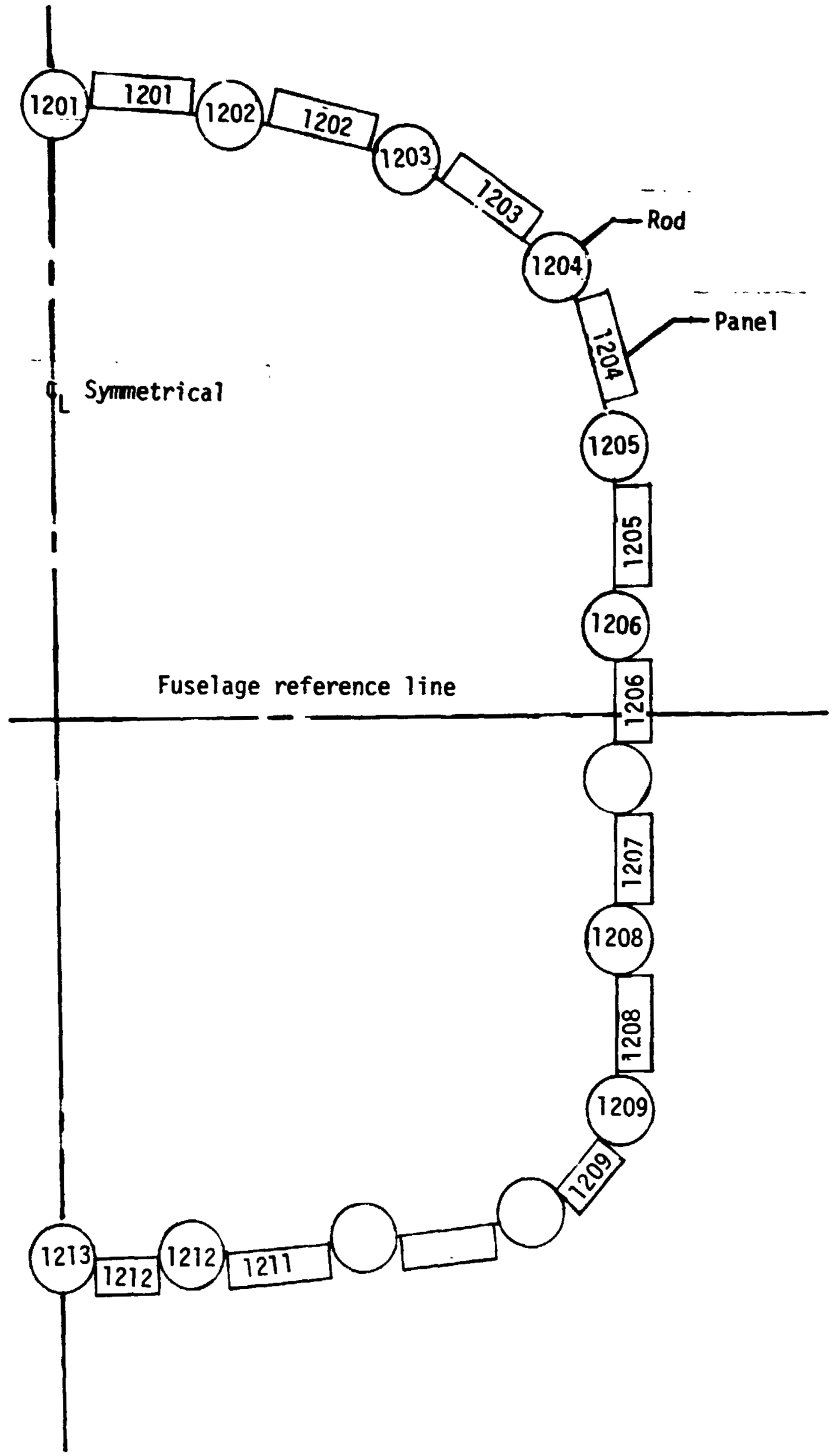


Figure 19.- NASTRAN model of typical payload bay section.

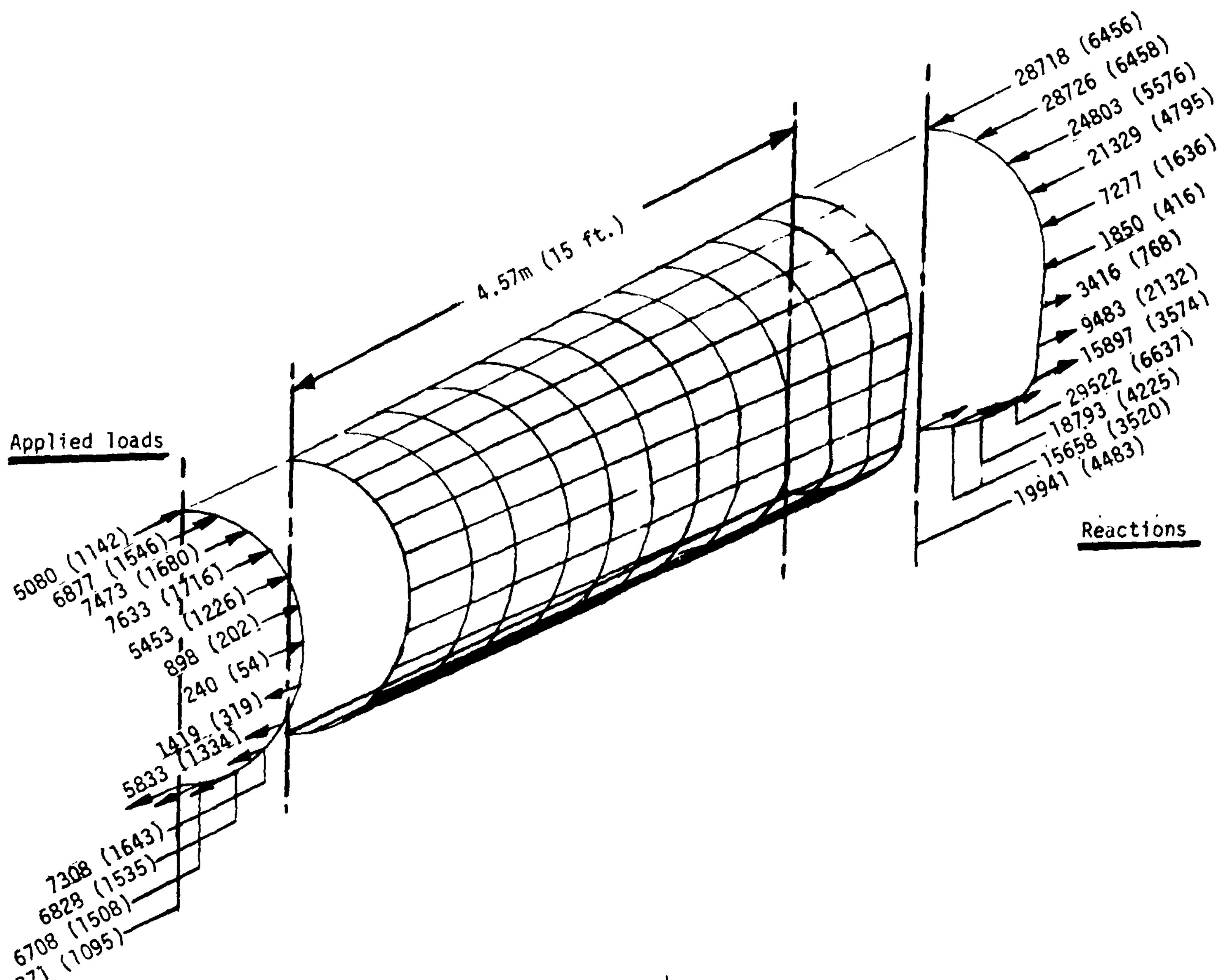


Figure 20.-NASTRAN derived axial loads and reactions at limit load for the payload bay (units are in N and (lb_f)).

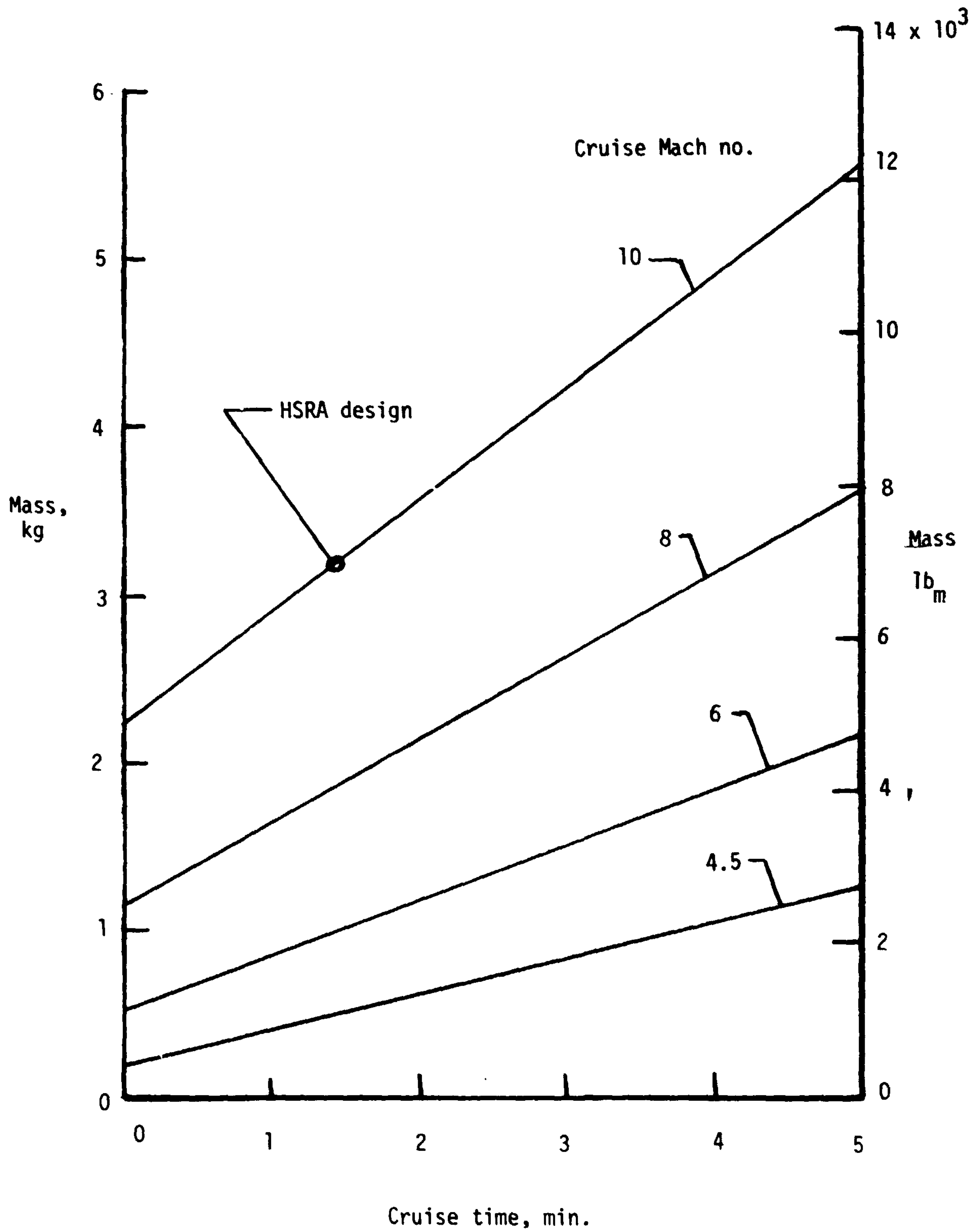


Figure 21.- Lockalloy mass for HSRA at a dynamic pressure of 23.9 kN/m^2 (500 psf) and at a temperature rise of 333°K (600°F).

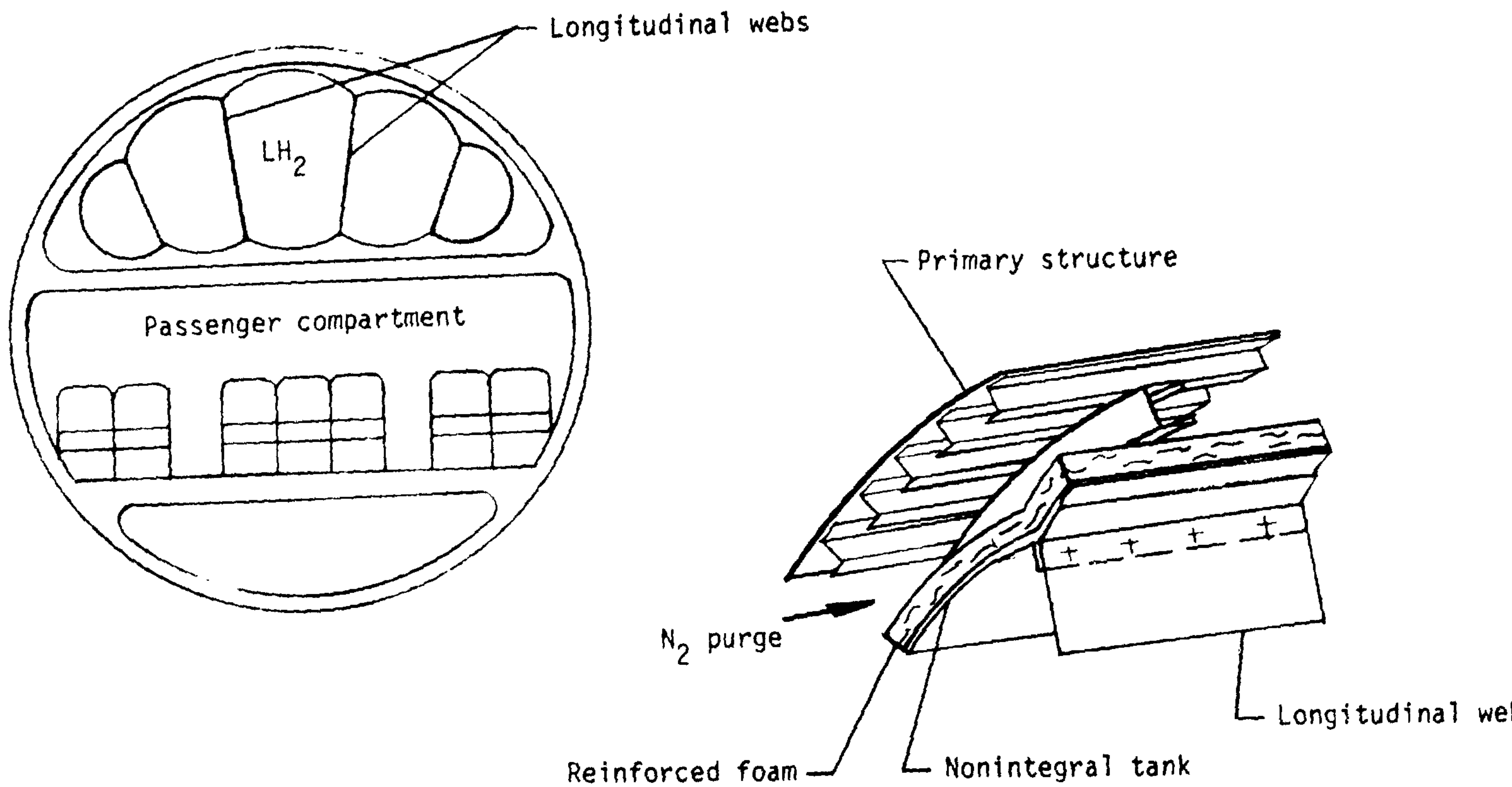
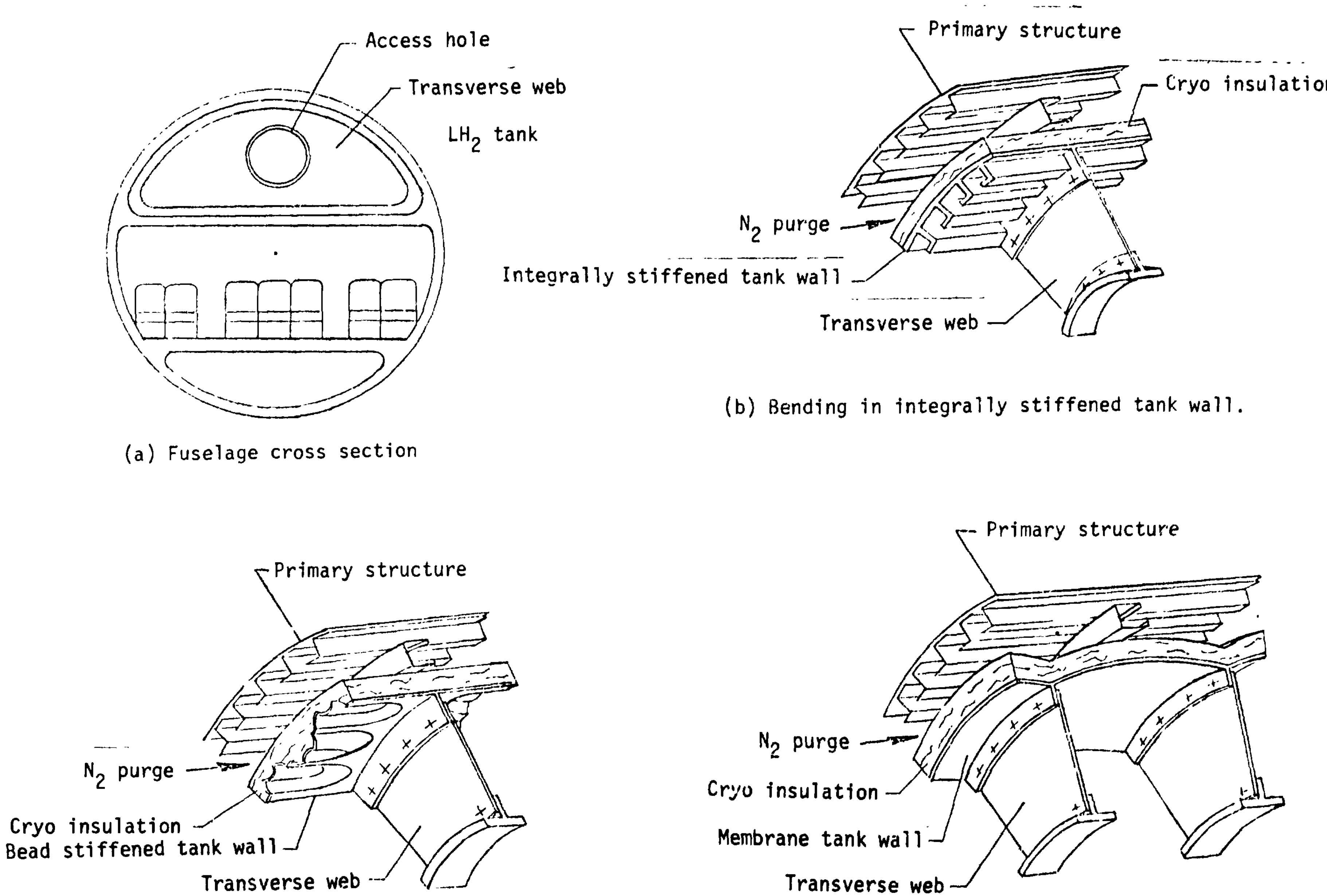


Figure 22.- Nonintegral multilobed liquid hydrogen tank.



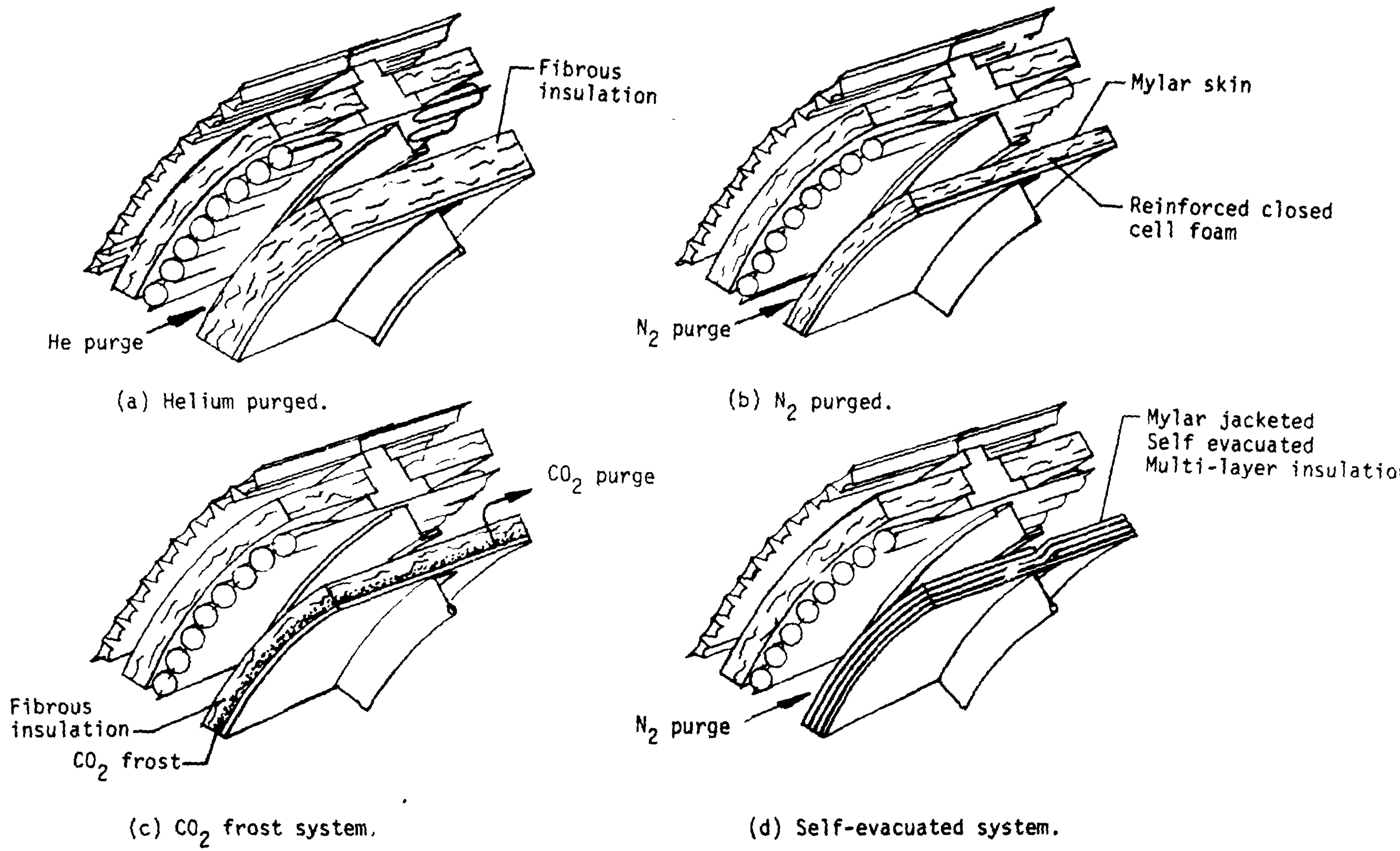
(a) Fuselage cross section

(b) Bending in integrally stiffened tank wall.

(c) Bending in bead stiffened tank wall

(d) Tension in membrane tank wall.

Figure 23.- Noncircular nonintegral liquid hydrogen tank designs.



(a) Helium purged.

(b) N_2 purged.

(c) CO_2 frost system.

(d) Self-evacuated system.

Figure 24.- Cryogenic thermal protection systems.

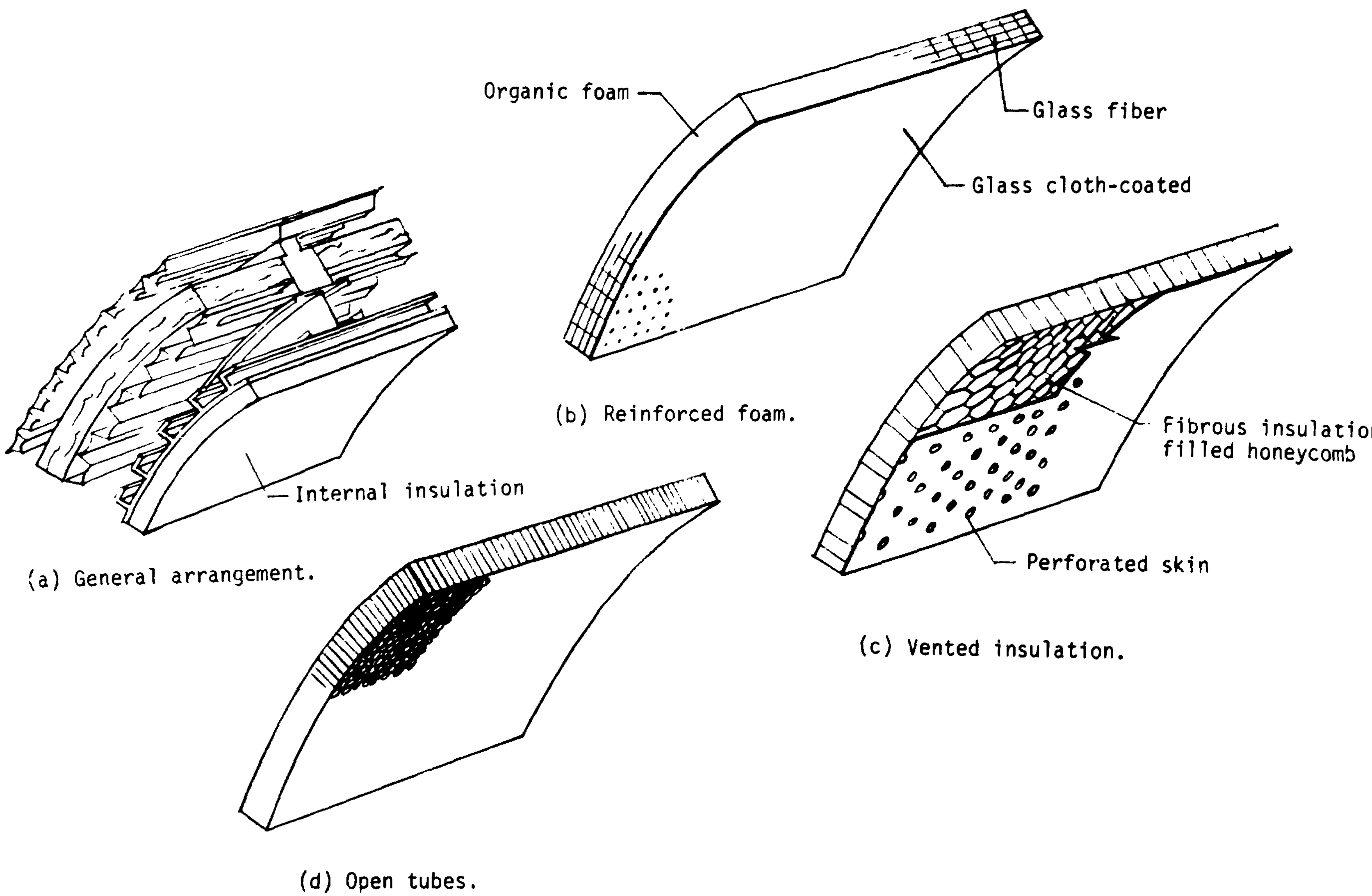


Figure 25.- Internal cryogenic insulations.

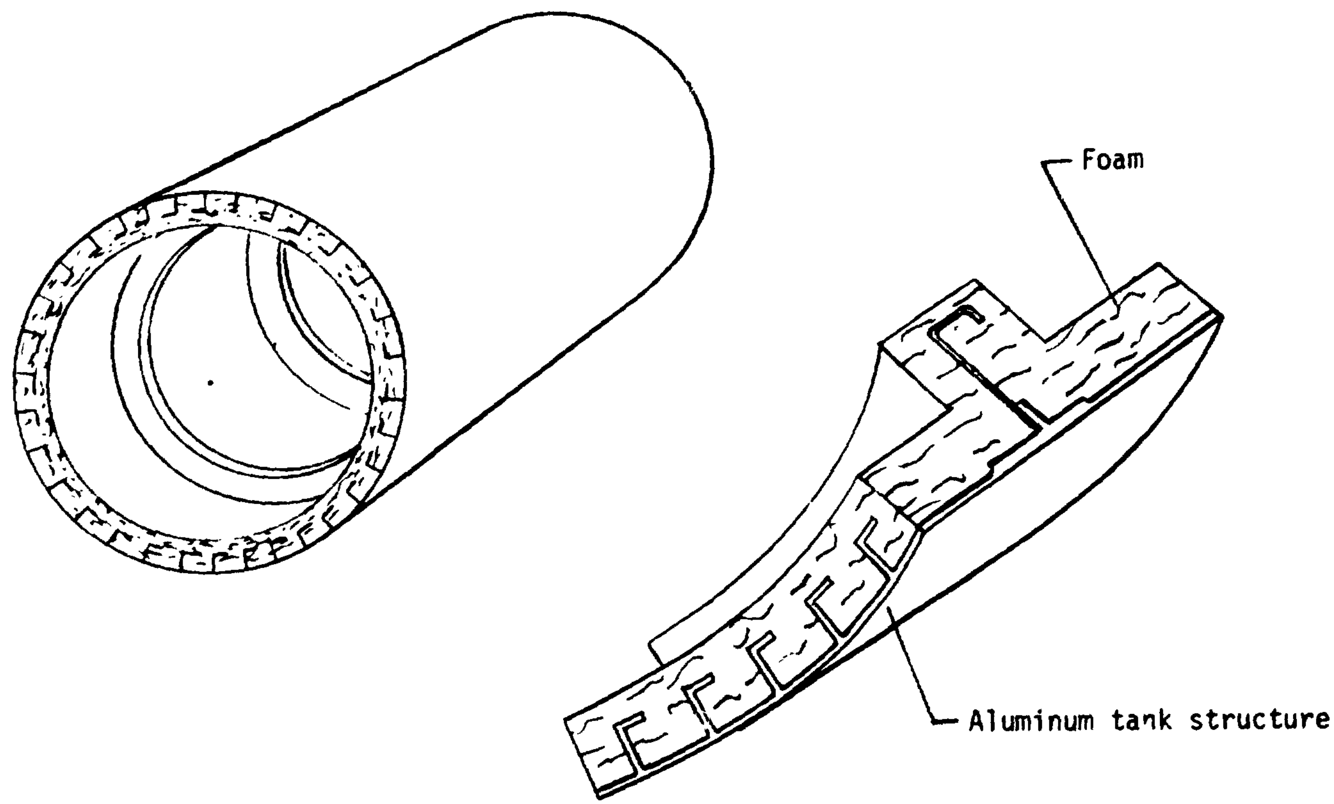


Figure 26.- Integral tank subsonic airplane.

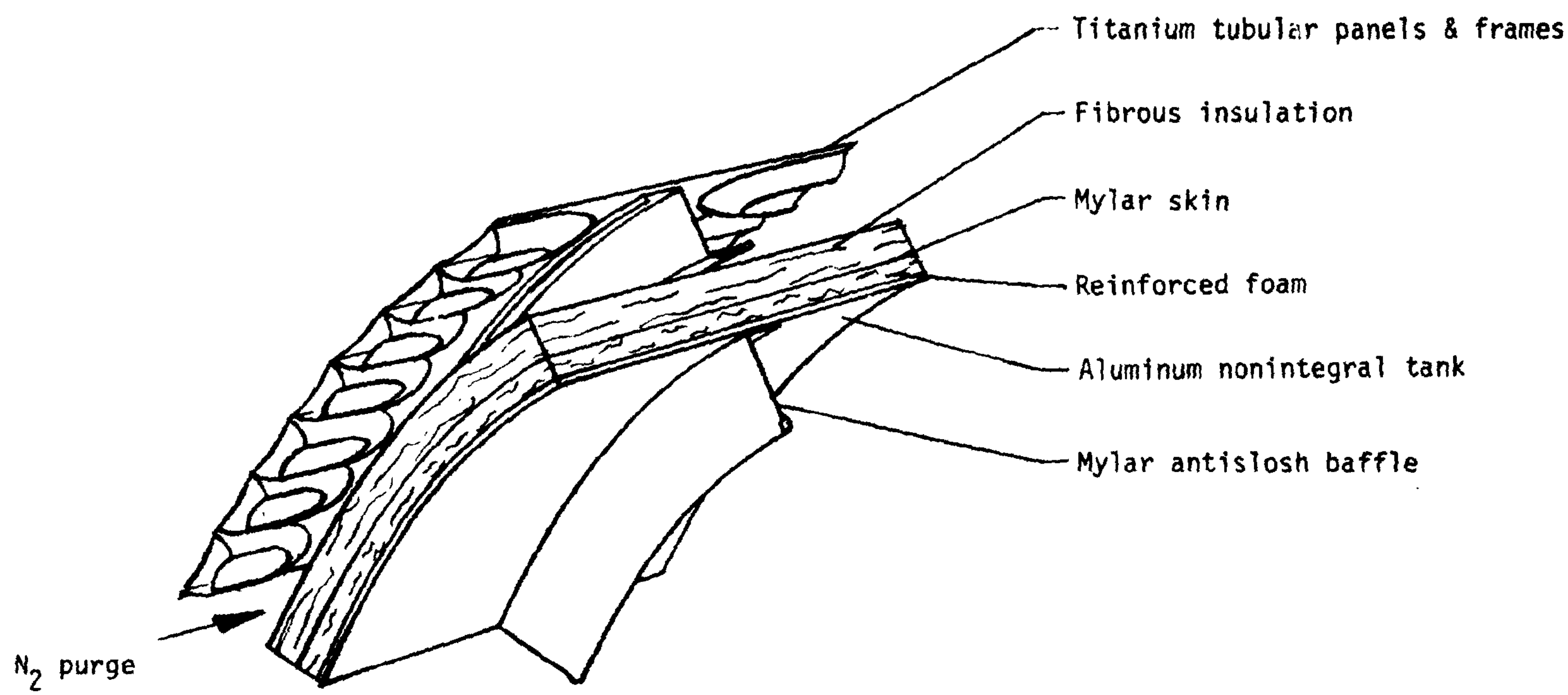


Figure 27.- Supersonic transport hot structure.

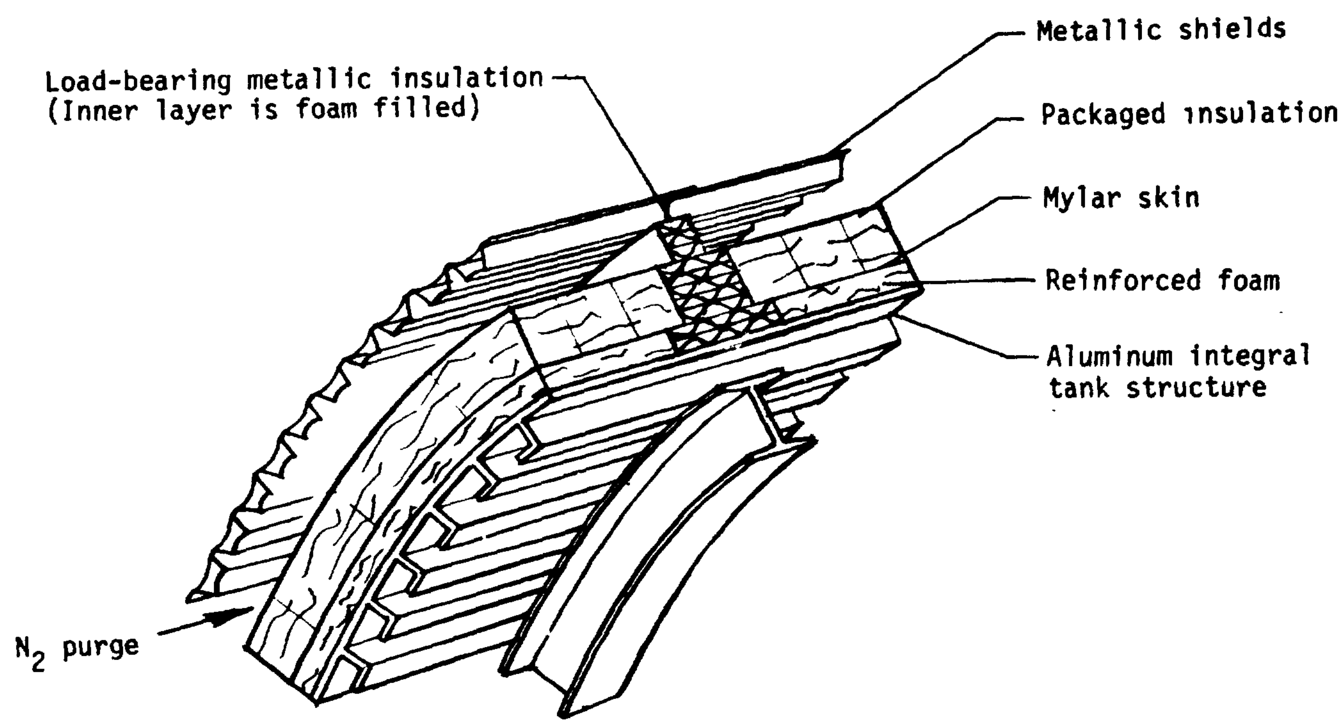


Figure 28.- Insulated structure-integral tank.

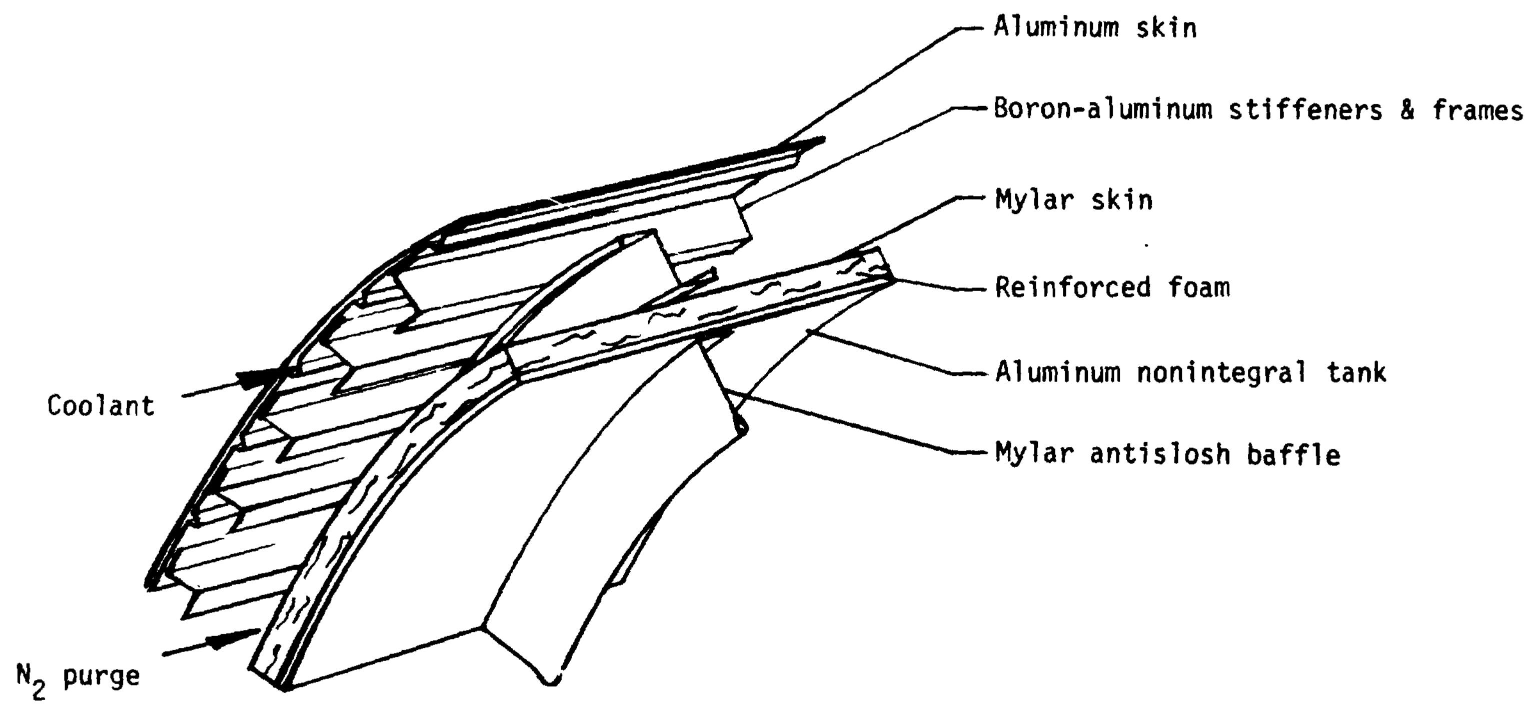


Figure 29.- Actively cooled structure.

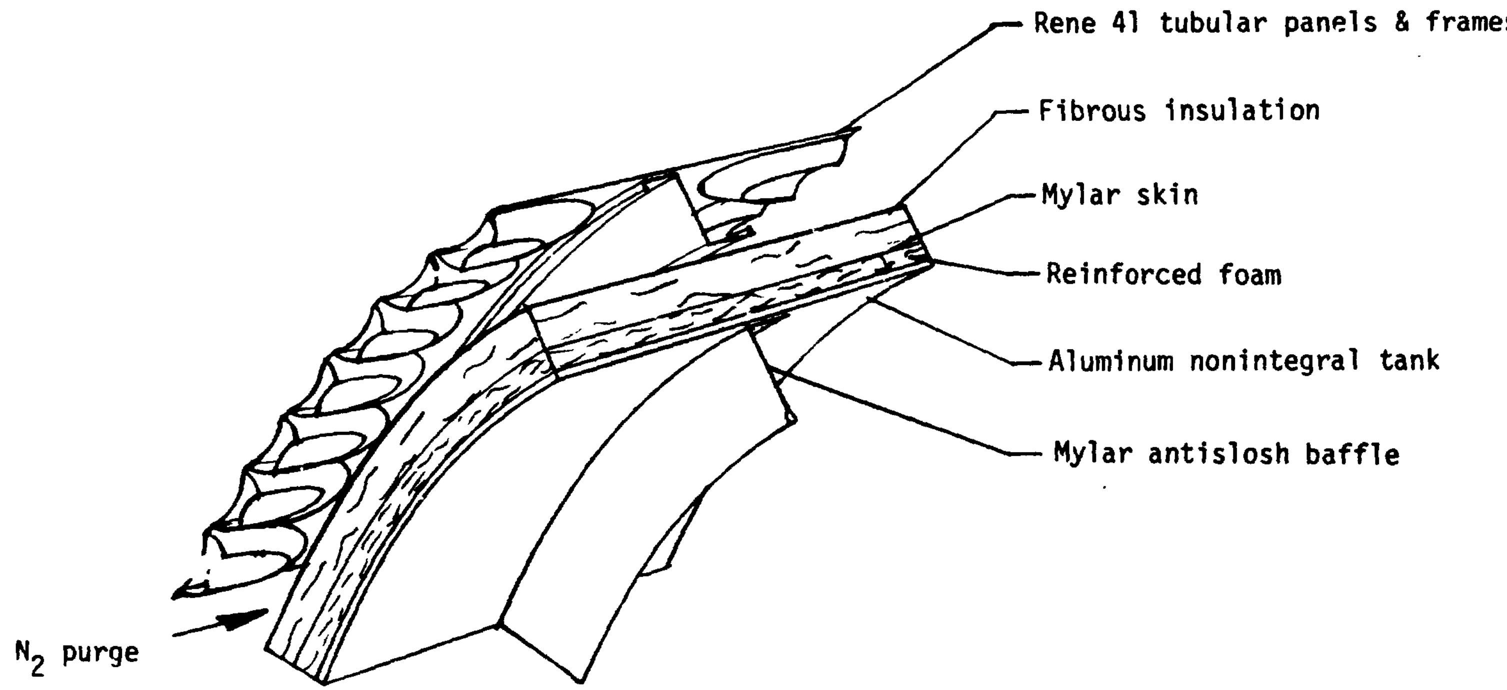


Figure 30.- Hypersonic transport hot structure.

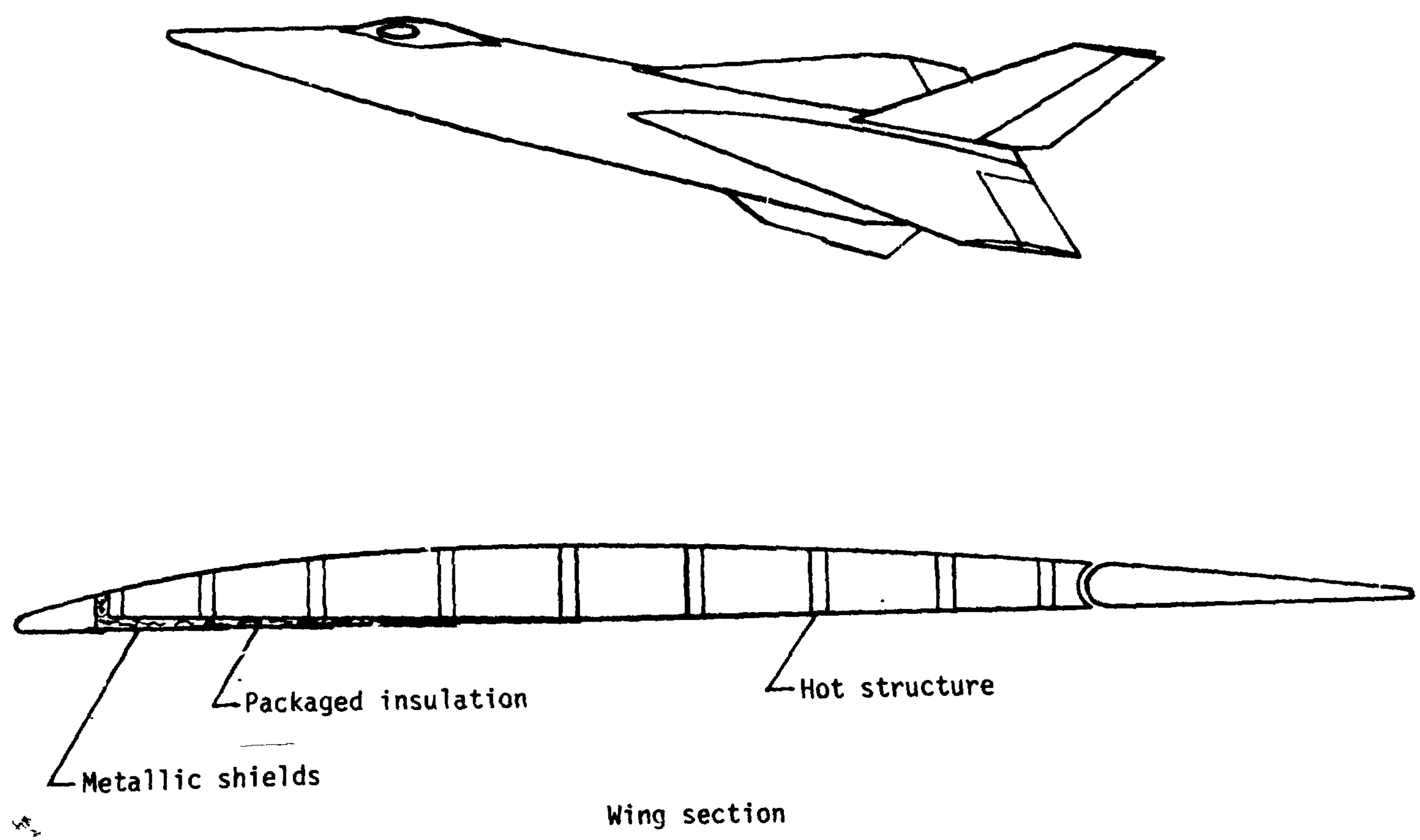


Figure 31.- Partially insulated structure.

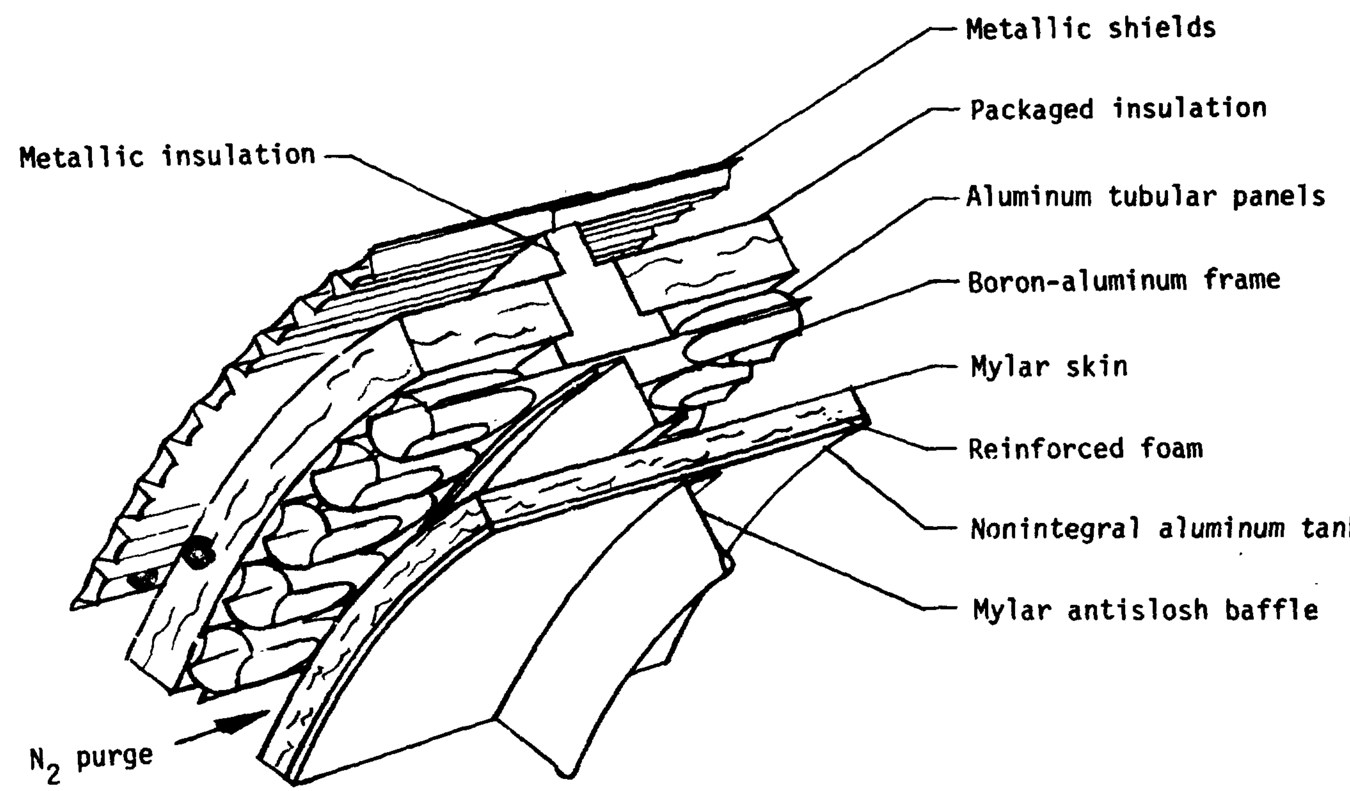


Figure 32.- Insulated structure with nonintegral tank.

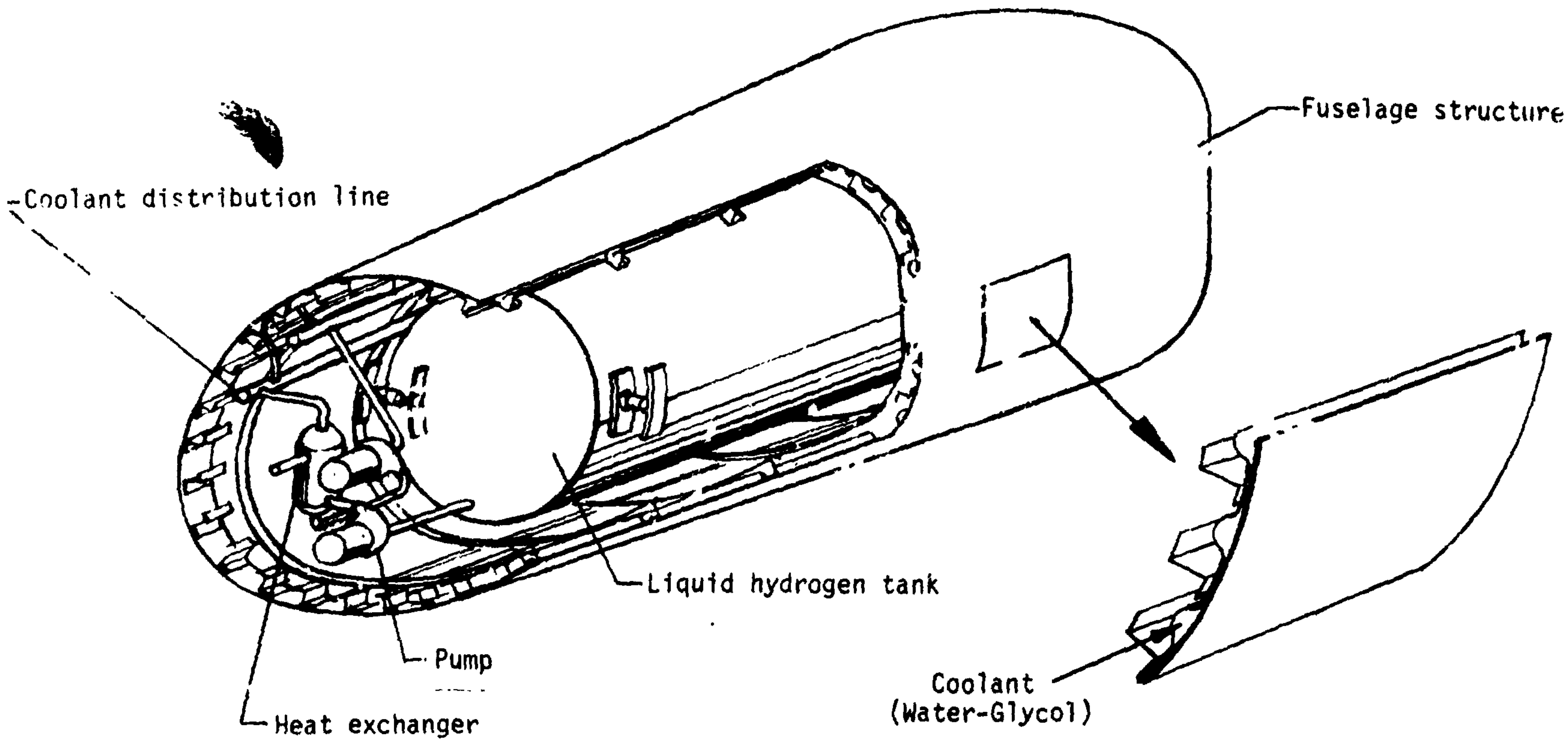


Figure 33.- Hypersonic t 1sport actively cooled structure.

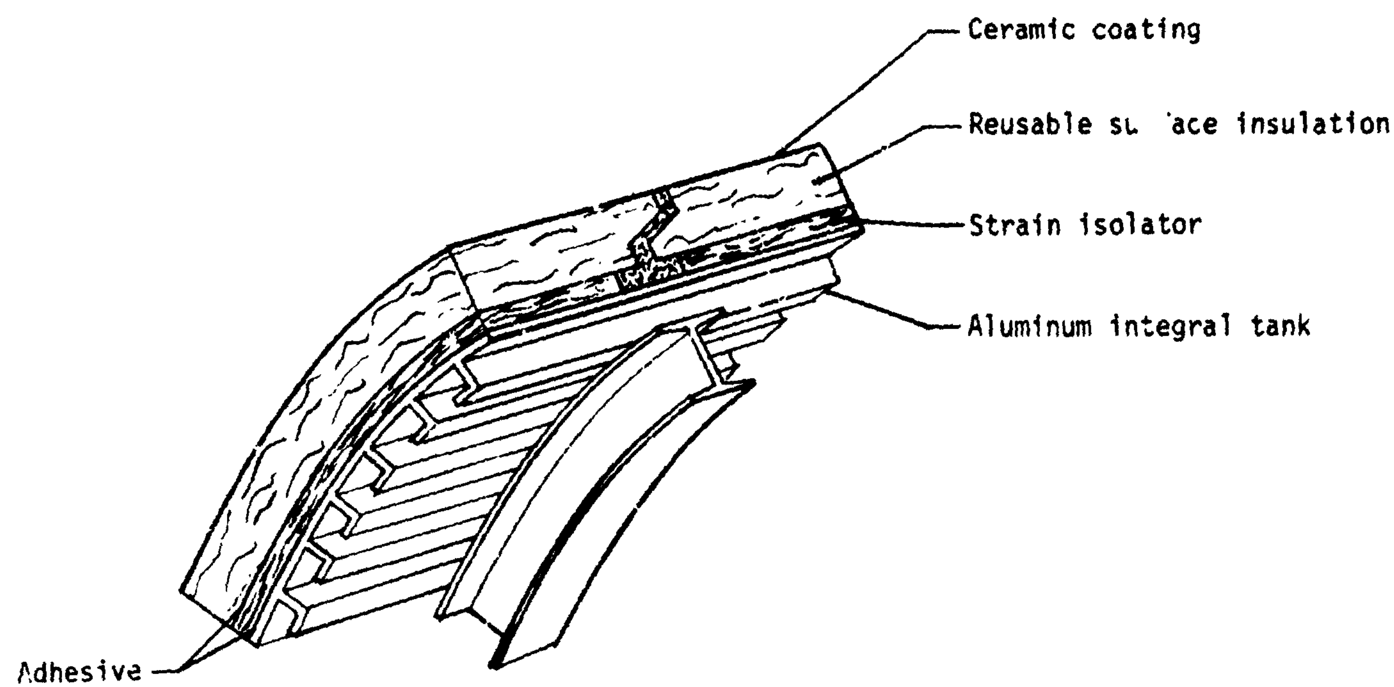
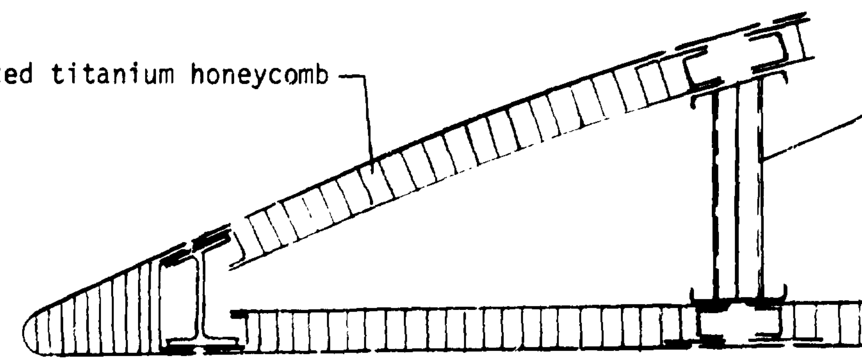
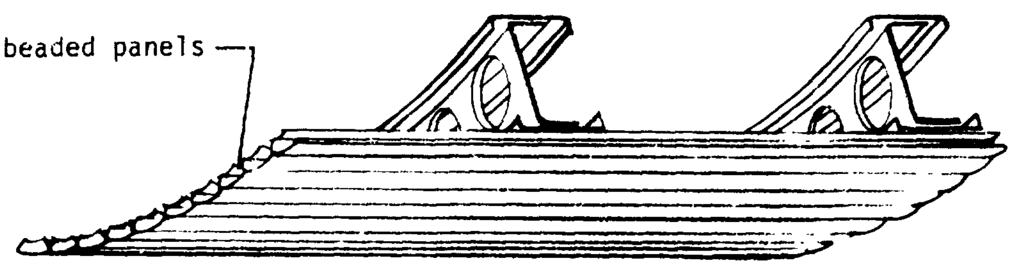


Figure 31.- Airbreathing launch vehicle structure protected with reusable surface insulation.



(a) Wing structure.



(b) Fuselage structure.

Figure 35.- Military airplane hot structure.

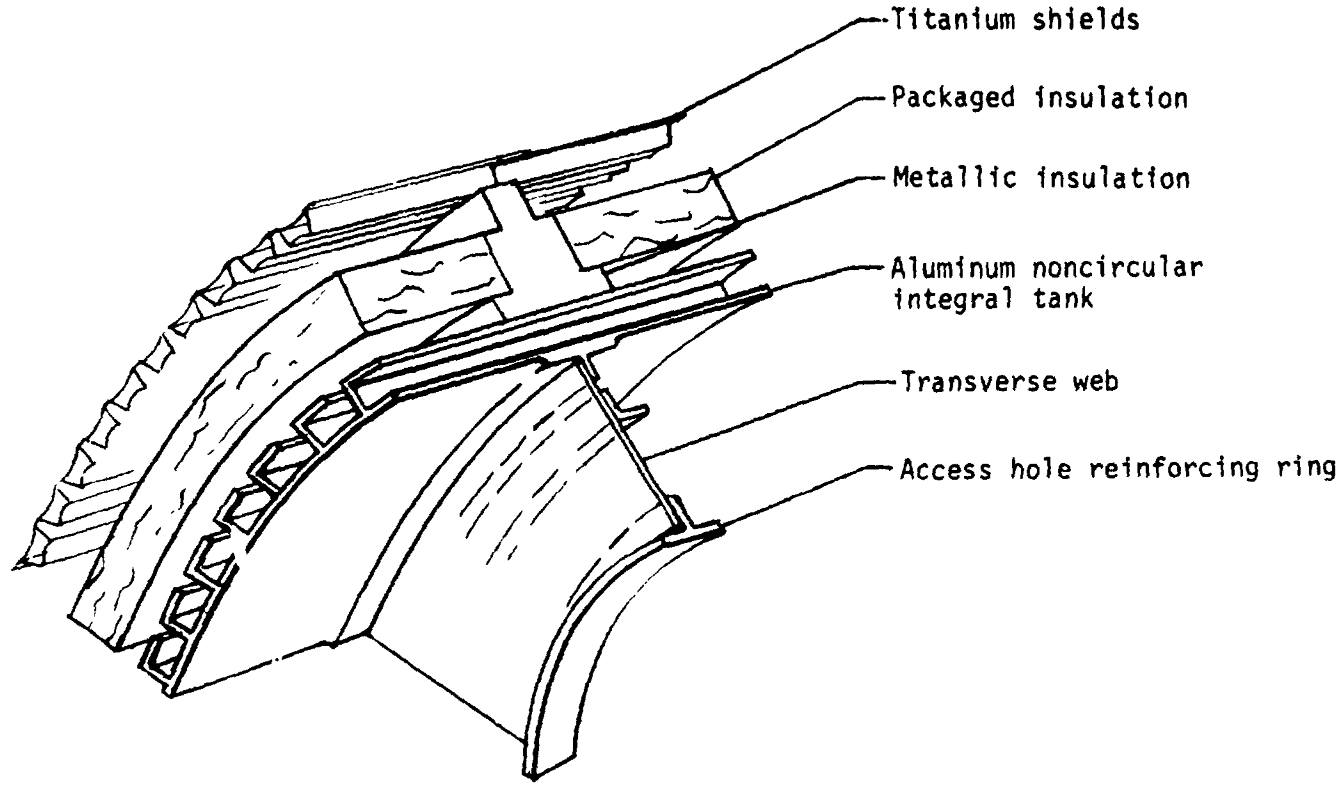


Figure 36.- Military airplane insulated structure with noncircular integral tank for JP fuel.

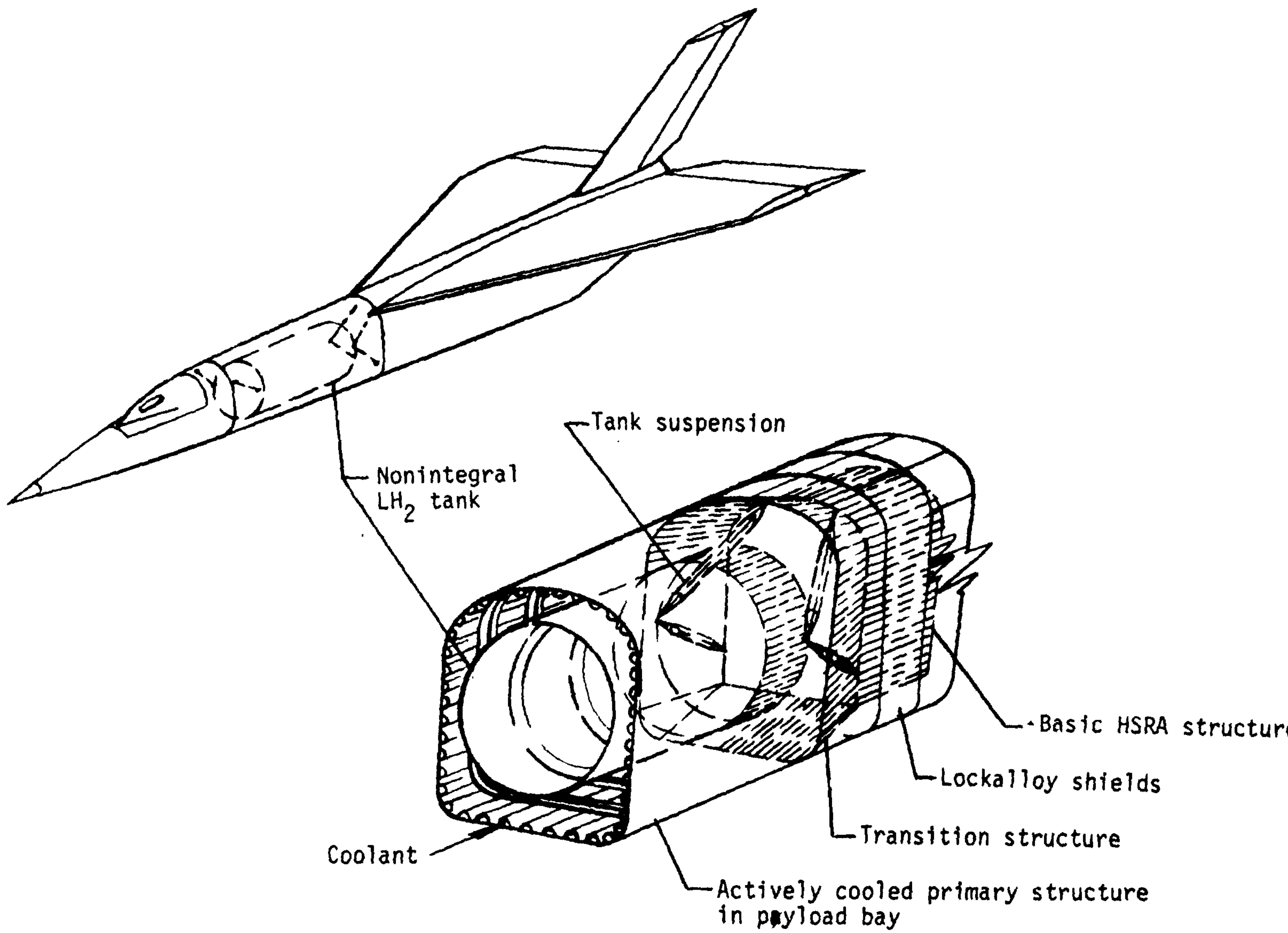


Figure 37.- Nonintegral LH_2 tank and actively cooled structure research payloads in HSRA.

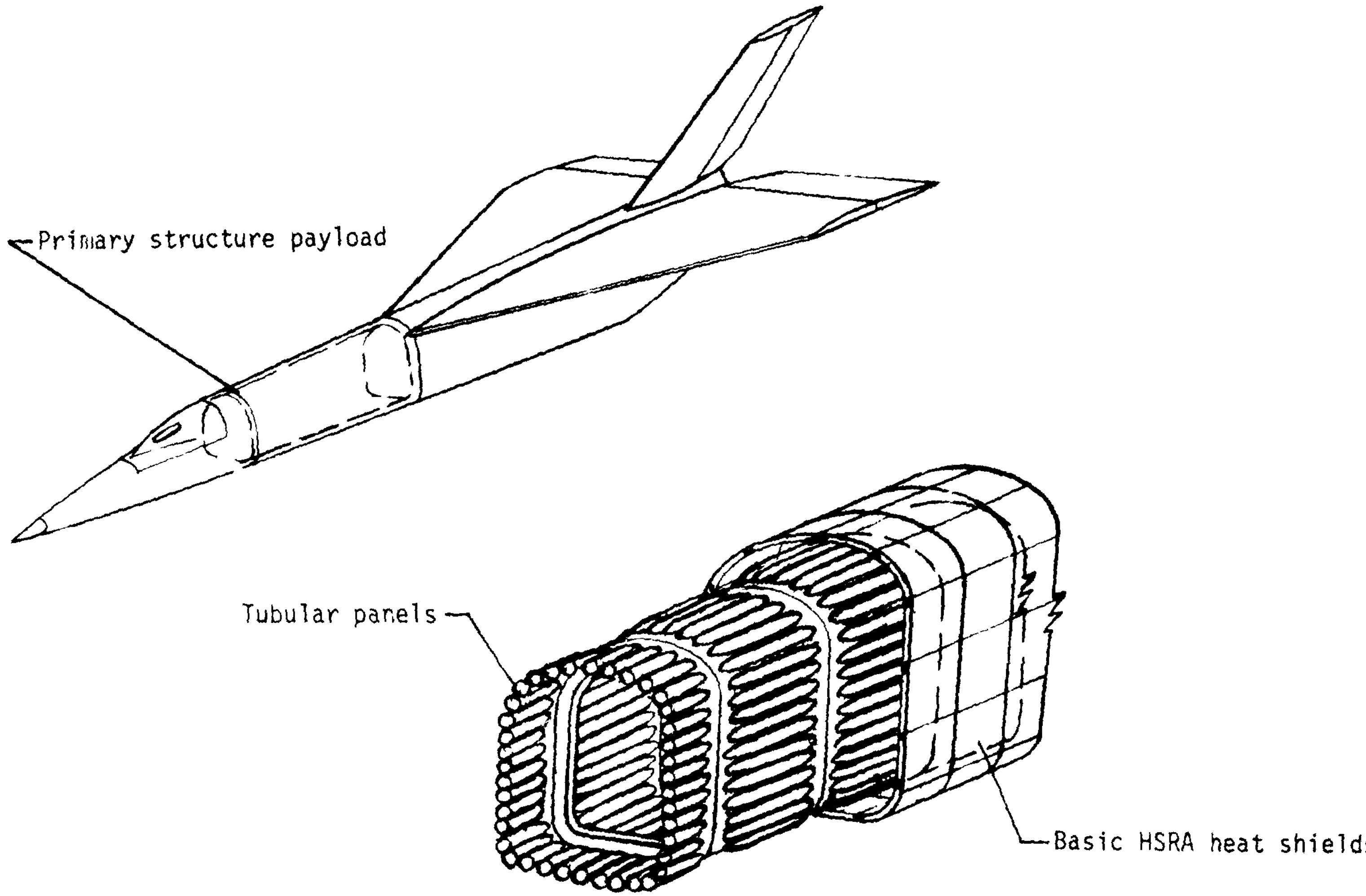


Figure 38.- Primary structure research payload in HSRA.

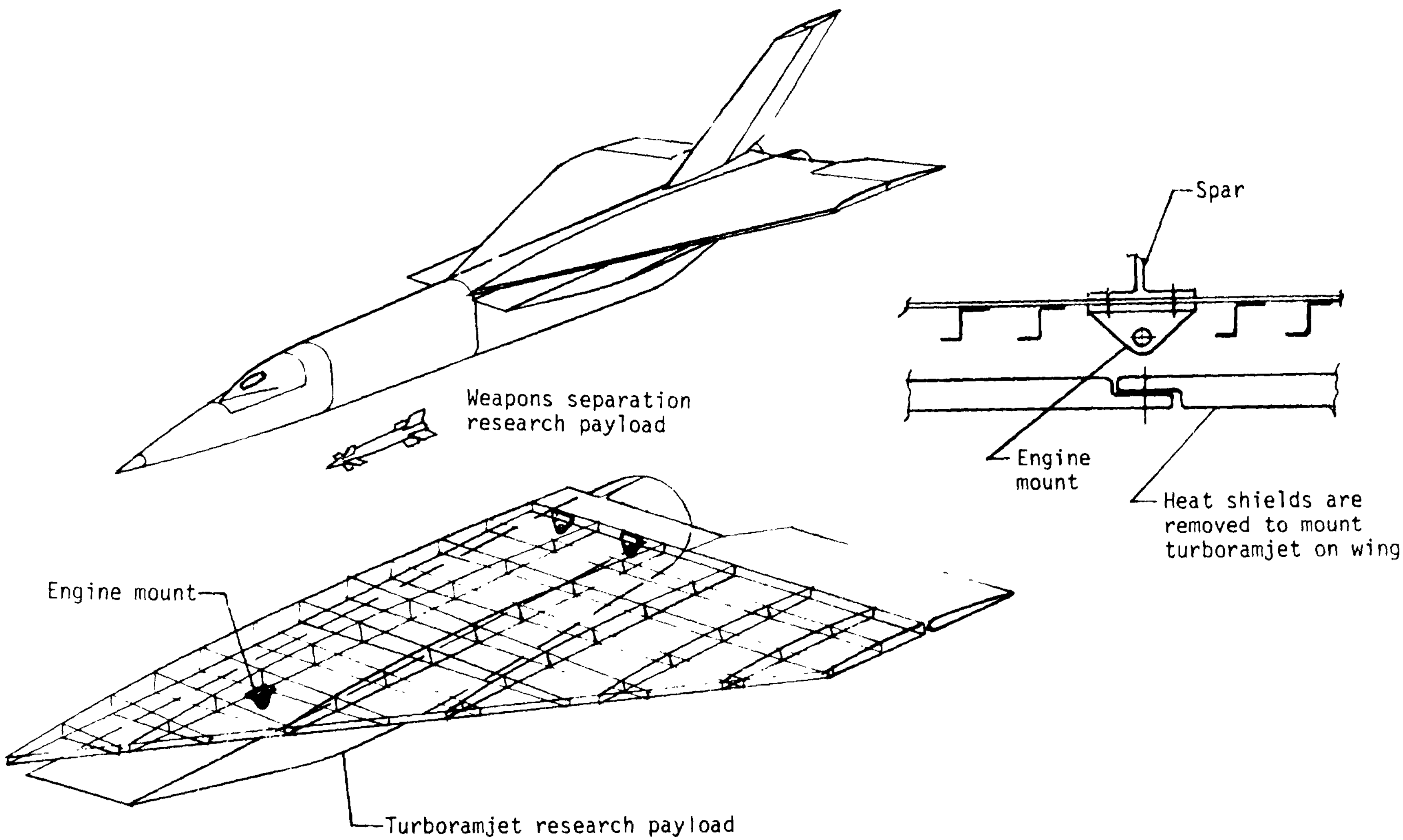


Figure 39.- Turboramjet and weapons separation payloads in HSRA.